



DUAL THROAT THRUSTER COLD FLOW ANALYSIS

Final Report

By

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16 Abstract The dual throat concept provides a means to obtain a large area ratio adjustment within a single thrust chamber assembly without the need for extendible nozzles. The concept was evaluated with cold flow (nitrogen gas) testing and through analysis for application as a tripropellant engine for single-stage-to-orbit type missions. Three modes of operation were tested and analyzed (1) Mode I Series Burn, (2) Mode I Parallel Burn, and (3) Mode II. Primary emphasis was placed on the Mode II plume attachment aerodynamics and performance. The conclusions from the test data analysis are as follows: (1) the concept is aerodynamically feasible, (2) the performance loss is as low as 0.5 percent, (3) the loss is minimized by an optimum nozzle spacing corresponding to an A_f/A_{fs} ratio of about 1.5 or an L_e/R_{tp} ratio of 3.0 for the dual throat hardware tested, requiring only 4% bleed flow, (4) the Mode I and Mode II geometry requirements are compatible and pose no significant design problems.					
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FOREWORD

The work described herein was performed at the Aerojet Liquid Rocket Company under NASA Contract NAS 8-32666 with Mr. Fred W. Braam, NASA-Marshall Space Flight Center, as Project Manager. The ALRC Program Manager was Mr. Larry B. Bassham, the Operations Project Manager was Dr. R. J. LaBotz, the Project Engineer was Mr. Charles J. O'Brien, the Analytical Design Engineer was Mr. Robert B. Lundgreen, and the Performance Analysis Consultant was Mr. Gary R. Nickerson.

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J. L. Pieper	Supervision, Analytical Design and Performance Analysis Effort
J. W. Salmon	Consultant, Analytical Design Effort
R. G. Spencer	Test Engineer

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NOMENCLATURE

<u>Term</u>	<u>Definition</u>	<u>Units</u>
A_F	Flow area between primary nozzle exit and secondary chamber wall as defined in Figure 31	M^2
A_{TS}	Secondary chamber throat area	M^2
Ca_∞	Crocco number defined in Section IV.B.3	
G	Mass flow entrained on the base flow side of the dividing streamline	Kg/s
G_T	Total mass flow	Kg/s
L_e	Length between primary nozzle exit and secondary chamber throat as defined in Figure 17	M
x_j	Distance along shear layer of plume	M
M_b	Mach number of the plume boundary (Figure 35)	-
M_e	Mach number at the exit of the primary nozzle	-
P_b	Static pressure of plume boundary	N/cm^2
P_{cp}, P_o	Primary chamber stagnation pressure	N/cm^2
P_{cs}	Secondary chamber pressure	N/cm^2
r	Radius of primary nozzle plume used to define plume shape in terms of r/r_e	M
r_e	Radius of primary nozzle exit used to define plume shape in terms of X/r_e	M
R_{tp}	Primary chamber throat radius	M
T_b	Static temperature of the plume boundary	$^{\circ}K$
T_o	Stagnation temperature of primary stream	$^{\circ}K$
X	Axial distance from primary nozzle exit used in defining plume shape	M
Y	Dimensional distance measured from the centerline of the shear layer	M

NOMENCLATURE (cont.)

<u>Term</u>	<u>Definition</u>	<u>Units</u>
γ	Expansion coefficient (ratio of heat capacities - C_p/C_v)	-
δ	Plume expansion angle, defined as the nozzle lip angle θ_n plus the Prandtl-Meyer expansion angle Δv in Section IV.B.2 (Figure 35)	degree
δ^*	Displacement thickness across the shear layer as defined in Section IV.B.3.c.	M
ϵ_p	Primary nozzle area ratio $(R_{ep}/R_{tp})^2$ as described in Figure 3	-
ϵ_s	Secondary nozzle area ratio $(R_{es}/R_{ts})^2$ as described in Figure 3	-
θ_E	Momentum thickness across the shear layer as defined in Section IV.B.3.c	M
σ	Shape factor of the shear layer	
ϕ	Velocity profile	

I. SUMMARY

A. STUDY OBJECTIVES AND SCOPE

The major objectives of this program were to: (1) evaluate the results of dual throat thruster cold flow tests, generated on a joint NASA-MSFC and ALRC effort, (2) conduct additional tests where required to complete the data matrix, (3) define the operating characteristics of the MSFC dual throat thruster concept, and (4) generate a computer model that adequately represents the dual throat aerodynamics and allows the prediction of performance for dual throat configurations.

To accomplish the objectives, a four task program, summarized in Figure 1, was conducted.

Test data were analyzed for a total of 69 tests conducted with the dual throat hardware supplied by the Marshall Space Flight Center. The first 40 tests were conducted under ALRC IR&D funding. The configurations tested on the IR&D program are shown in Figure 2. An additional 29 Mode II plume attachment tests were completed on this contract. The Mode II hardware configurations tested are illustrated in Figure 3.

An aerodynamic mixing model was formulated to correlate the test data. The logic used to develop the aerodynamic model and to incorporate it into a performance model for dual-throat engine performance is illustrated in Figure 4. The cold flow test data was used to gain basic understanding of dual throat operating characteristics and to identify the most important design concept variables. The data were also used to identify the influencing physical mechanisms such as shear mixing, plume shape and impingement, weak compression shocks, and secondary wall two-dimensional expansion. Identification of the important physical mechanisms led to selection of analytical methods of problem solution.

The methodology was established for predicting the performance of dual throat thrusters and for making a preliminary performance prediction for a hot firing type test. This methodology incorporated the aerodynamic model and utilized existing simplified JANNAF models for performance prediction.

A preliminary engine system model was defined utilizing the design criteria generated in this program.

B. RESULTS AND CONCLUSIONS

The most important conclusion from this study is that the dual throat concept is aerodynamically feasible. The performance loss obtained during cold flow was as low as 0.5 percent. The loss is minimized by an optimum nozzle spacing corresponding to an A_f/A_{TS} ratio of about 1.5, or an

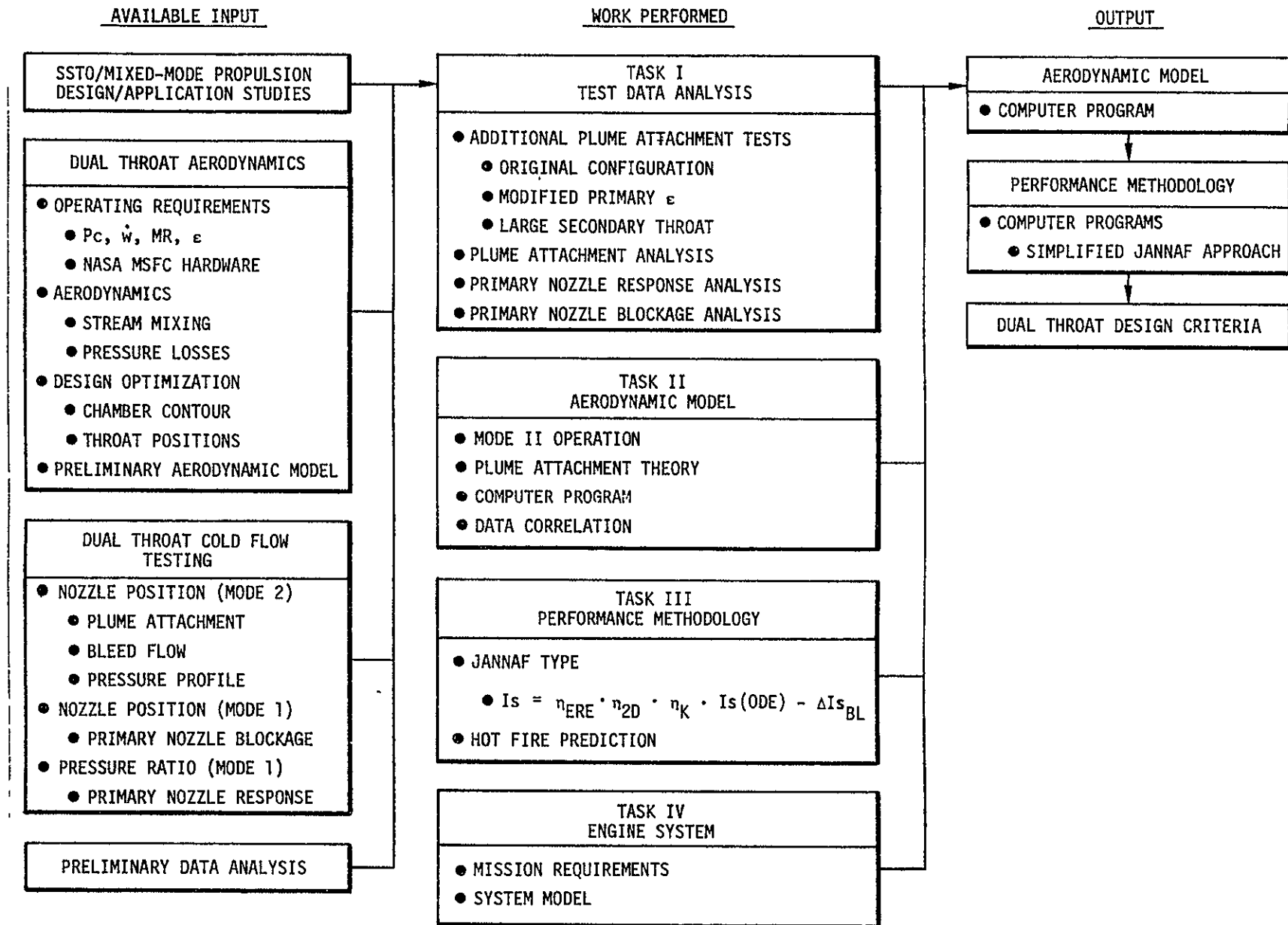
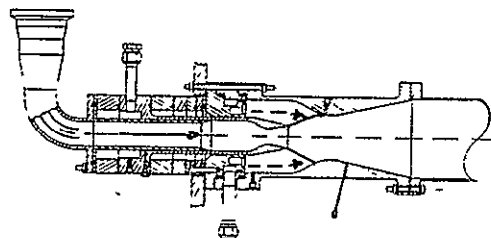


Figure 1. Dual Throat Thruster Cold Flow Analysis Program Logic

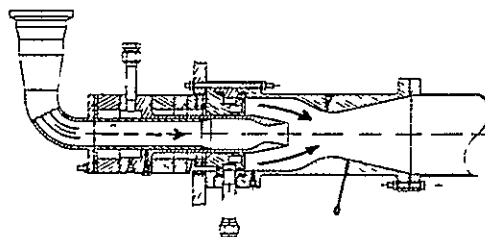
● MODE II
PLUME ATTACHMENT TESTS



—▶ HIGH PRIMARY FLOW
--▶ LOW BLEED FLOW

- DEMONSTRATE DUAL THROAT AREA RATIO ADJUSTMENT
- MEASURE DUAL THROAT PERFORMANCE LOSSES
- EVALUATE INFLUENCE OF BLEED FLOW AND PRIMARY NOZZLE POSITION ON PERFORMANCE
- CORRELATE TEST DATA TO FORMULATE A MODE II PERFORMANCE MODEL

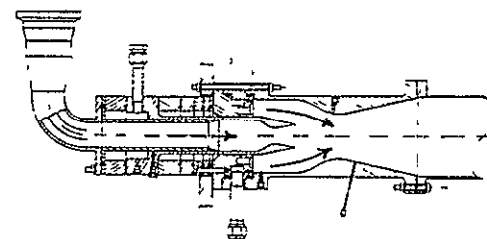
● MODE I SERIES BURN
SECONDARY BLOCKAGE TESTS



—▶ HIGH SECONDARY FLOW
--▶ ZERO PRIMARY

- DETERMINE INFLUENCE OF PRIMARY NOZZLE POSITION ON MODE I SERIES BURN OPERATION
- EVALUATE EXPANSION LOSSES

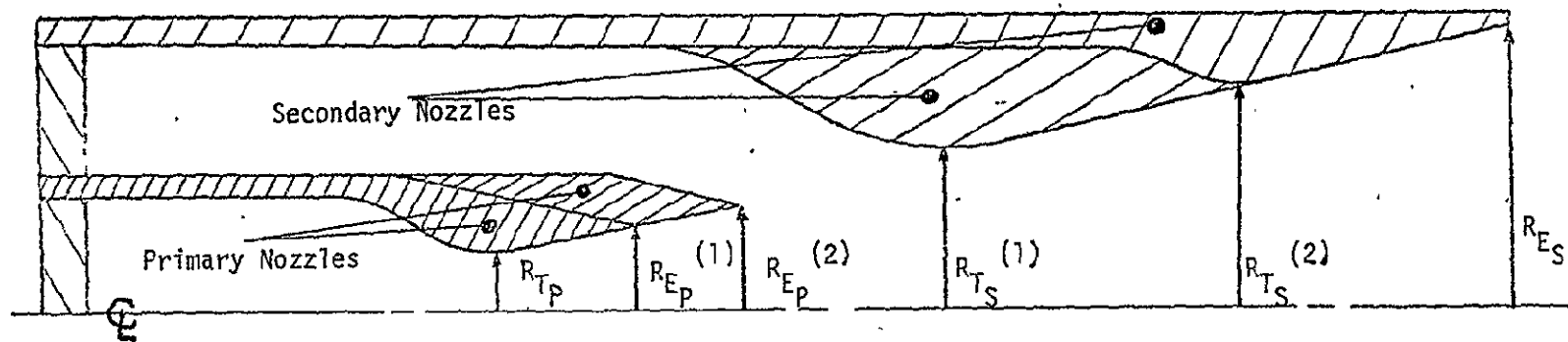
● MODE I - PARALLEL BURN
PARALLEL FLOW RESPONSE TESTS



—▶ HIGH SECONDARY FLOW
--▶ HIGH PRIMARY FLOW

- DEMONSTRATE FEASIBILITY OF PRIMARY SONIC FLOW WITH RELATIVE HIGH BACK PRESSURE
- DETERMINE RANGE OF FLOW SPLITS AND CHAMBER PRESSURE RATIOS ASSOCIATED WITH SONIC FLOW IN THE PRIMARY THROAT

Figure 2. Dual Throat Test Configurations



HARDWARE CONFIGURATIONS [cm (in.)]

$$R_{Tp} = 1.739 (.6845) \quad R_{Es} = 7.874 (3.100) \quad \epsilon_T = \left(\frac{R_{Es}}{R_{Tp}} \right)^2 = 20$$

TEST NUMBER	PRIMARY NOZZLE		SECONDARY NOZZLE	
	R_{Ep}	ϵ_p	R_{Ts}	ϵ_s
101A - 110	2.244 (.8835)	1.666 ⁽¹⁾	4.562 (1.796)	2.979 ⁽¹⁾
111 - 116	2.244 (.8835)	1.666 ⁽¹⁾	6.175 (2.431)	1.626 ⁽²⁾
117 - 124	3.092 (1.218)	3.164 ⁽²⁾	6.175 (2.431)	1.626 ⁽²⁾
125 - 130	3.092 (1.218)	3.164 ⁽²⁾	4.562 (1.796)	2.979 ⁽¹⁾

Figure 3. MSFC Contractual Test Series, Dual Throat Cold Flow
Mode II Hardware Configurations

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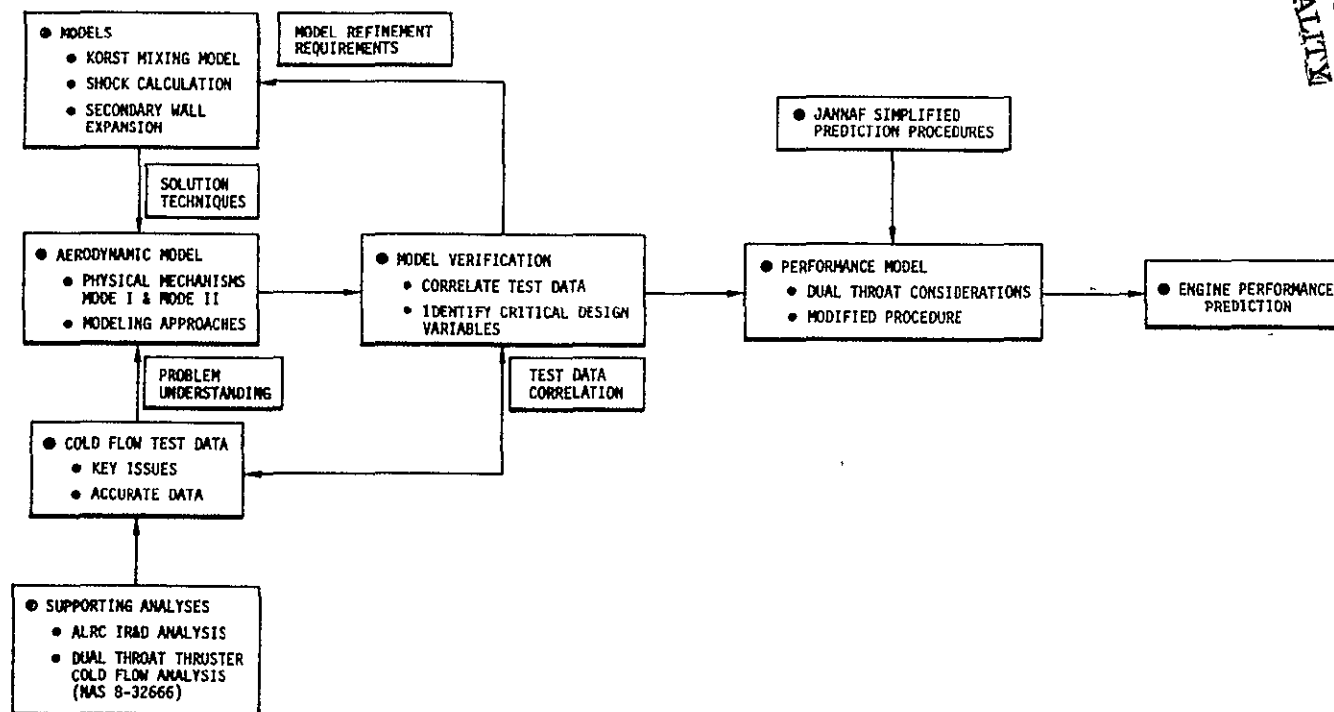


Figure 4. Dual Throat Performance Model Development

I, B, Results and Conclusions (cont.)

Le/R_{tp} ratio of 3 for the dual-throat hardware tested. Bleed flow requirement at this condition is only 4 percent. The results indicate that the Mode I and Mode II geometry requirements are compatible and pose no significant design problems.

C. CONTRIBUTION TO THE DEFINITION OF HIGH PERFORMANCE ENGINES

This study has been successful in confirming, from a performance viewpoint, that the dual throat thruster concept can be competitive with other tripropellant and mixed mode engines being considered for single-stage-to-orbit (SSTO) missions. It remains for these study results to be used in an engine system analysis to determine the performance/weight trade-offs and the vehicle/engine integration requirements.

II. INTRODUCTION

A. BACKGROUND

Based on both Apollo and Space Shuttle experience, the lead time from an "Authority to Proceed" to an operational system is of the order of eight to ten years, and the planning for, and the development of the necessary technology base for such programs precedes this time by ten to fifteen years. Utilizing this knowledge, NASA has made the assumption that a follow-on to the present Space Shuttle system will be required in the 1995 time frame, and since 1973 has conducted studies directed toward the examination of potentially attractive earth-orbit transportation systems as a means of identifying the associated technology requirements.

Advanced high pressure bipropellant engine and tri-propellant engine (TPE) concepts have emerged from these studies. These propulsion systems meet the unique requirement of high performance density demanded by the single-stage-to-orbit and heavy lift launch vehicle concepts, and thus offer a means to achieve an economical space transportation system.

One of the major propulsion system candidates that is being studied is the dual throat concept, proposed earlier by NASA-MSFC to obtain a large area ratio adjustment within a single thrust chamber assembly, without the need for extendible nozzles. The concept is readily adaptable to TPE applications, and preliminary analysis has indicated a high nozzle efficiency.

B. PURPOSE AND SCOPE

The feasibility of an economical space transportation system is heavily dependent on the delivered performance of the engine system. The first step in evaluating a new engine concept for this application, therefore, is to determine the performance. It is the purpose of this study to correlate the results of nozzle cold flow tests conducted with a NASA-MSFC designed dual-throat thruster at the Aerojet Liquid Rocket Company, and to provide a means for determining the performance of a dual-throat engine.

C. GENERAL REQUIREMENTS

For purposes of this program both parallel burn and series burn conditions were assumed as requirements for the engine during Mode I operation. Mode I of the mixed-mode propulsion concept involves the sequential or parallel use of high density impulse propellants and high specific impulse propellants in a single stage to increase vehicle performance and reduce vehicle weight. The concept has been described in the literature (e.g., see Reference 26) and theoretically provides the performance necessary for a VTOHL-SSTO Space Transportation Vehicle.

II, C, General Requirements (cont.)

Mode II operation is assumed to involve only the high specific impulse propellant.

D. APPROACH

To accomplish the program objectives, an effort involving four technical tasks was conducted. Tasks accomplished were:

1. Task I: Dual Throat Cold Flow Test Data

Additional tests were conducted and the nozzle cold flow test data were analyzed to determine the performance potential of the dual-throat thruster.

2. Task II: Aerodynamic Mixing Model

Significant theories concerning plume shape and plume attachment were reviewed and compared with the results of the cold flow test data. An aerodynamic mixing model was formulated, a computer program was written, and the program was utilized to correlate the test data.

3. Task III: Performance Methodology

A method for predicting the performance of dual-throat thrusters was established and a preliminary performance prediction for a hot firing type test was made.

4. Task IV: Preliminary Engine System Model

A preliminary engine system model was defined utilizing the design criteria generated on this program.

III. DUAL-THROAT COLD FLOW TEST DATA

A. OBJECTIVES AND GUIDELINES

A total of 69 tests were conducted with the dual throat hardware supplied by the Marshall Space Flight Center. The tests were divided into three series: (1) plume attachment; (2) secondary only flow, and (3) parallel flow. A pictorial description of the tests and corresponding objectives are shown in Figure 2. Dual throat terminology is illustrated in Figure 5, which also includes a comparison between typical high pressure TPE design parameters and those for the dual throat cold flow test hardware.

The major objectives of this study are to: (1) substantiate previous analytical results, (2) provide data concerning dual throat performance, and (3) correlate the test results to actual engine design conditions.

B. EXPERIMENTAL APPARATUS

1. Dual Throat Assembly

The dual throat assembly supplied by Marshall Space Flight Center consists of two concentric nozzles, spacers for changing their relative axial positions, an injector for hot firing the inner nozzle, and manifold for injecting bleed flow into the annulus between the inner (primary) and the outer (secondary) nozzles. The secondary nozzle is equipped with 48 pressure taps in 21 axial positions. Thirty-six of the 48 available pressure taps were used during this test program.

The Dual Throat Assembly was modified to allow the high volume flows required for cold flow testing with gaseous nitrogen. To increase the flow capacity of the primary circuit the injector was removed; a new end cap with a two inch diameter inlet pipe was fabricated; and a sleeve was installed between the end cap and the chamber to fair the inside diameter of the inlet pipe to the inside diameter of the chamber. The capacity of the secondary circuit was increased by removing the diffuser ring, drilling a 1.031 cm (.406 in.) diameter hole between each of the .3175 cm (.125 in.) diameter holes in the secondary manifold, and replacing the four 1.588 cm (.625 in.) tubes on the secondary manifold cover with six 2.54 cm (one-inch) diameter tubes. Figure 6 shows the configuration of the dual throat assembly as it was tested.

The primary nozzle was modified late in the program to allow testing with two primary nozzle area ratios, and to verify the back pressure measurement obtained initially with secondary wall pressure measurements (cf. Figures 7 and 8). Six additional pressure taps were added to the primary nozzle outer wall. These measurements agreed well with the secondary wall measurements, substantiating the early test data.

● OPERATING MODES

MODES	PRIMARY	SECONDARY
I - SEA LEVEL SERIES BURN	0	100%
I - SEA LEVEL PARALLEL BURN	~20-30%	~70-80%
II - ALTITUDE	~90-100%	0-10%

● DESIGN PARAMETERS

DESIGN PARAMETER	TPE DESIGNS	DUAL THROAT HARDWARE
MODE I TO MODE II THRUST RATIO	2-6	5
SECONDARY AREA RATIO	45:1	3:1
PRIMARY AREA RATIO (C_p)	200-400:1	20:1
ϵ_p/ϵ_s	4-9	7
CHAMBER PRESSURE N/CM ² (PSIA)	1379-4137 (2000-6000)	207 (300)
EXIT PRESSURE N/CM ² (PSIA)	0.14-0.69 (0.2-1.0)	1.0 (1.5)

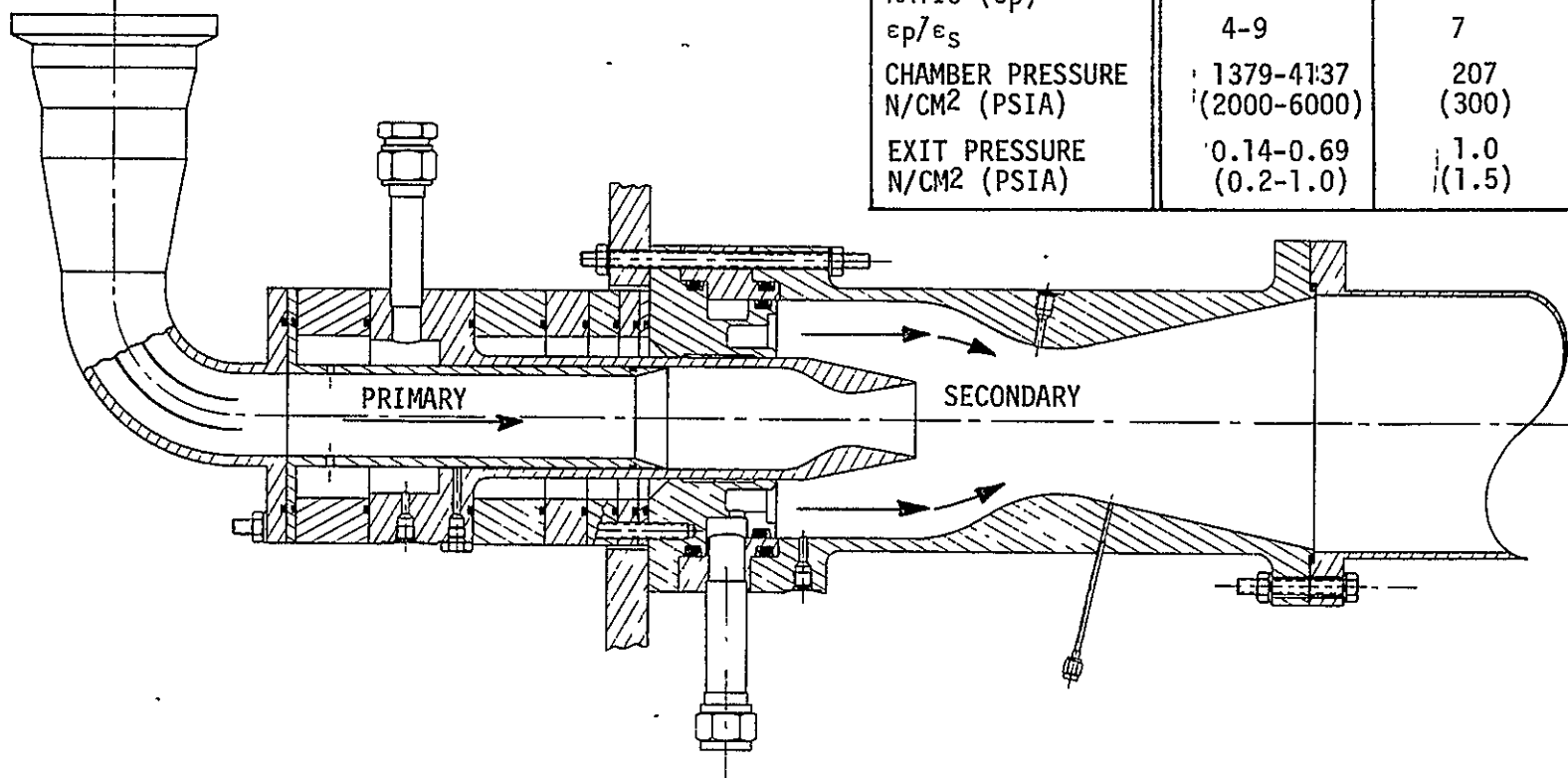


Figure 5. Dual Throat Terminology

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- DIRECT MEASUREMENT (USED IN ALL THREE TEST SERIES)
 - PRESSURE - SECONDARY WALL, PRIMARY P_C , SECONDARY P_C
 - TEMPERATURE - PRIMARY AND SECONDARY INLET
 - FLOW RATE - SONIC NOZZLE + REQUIRED PRESSURES & TEMPERATURES

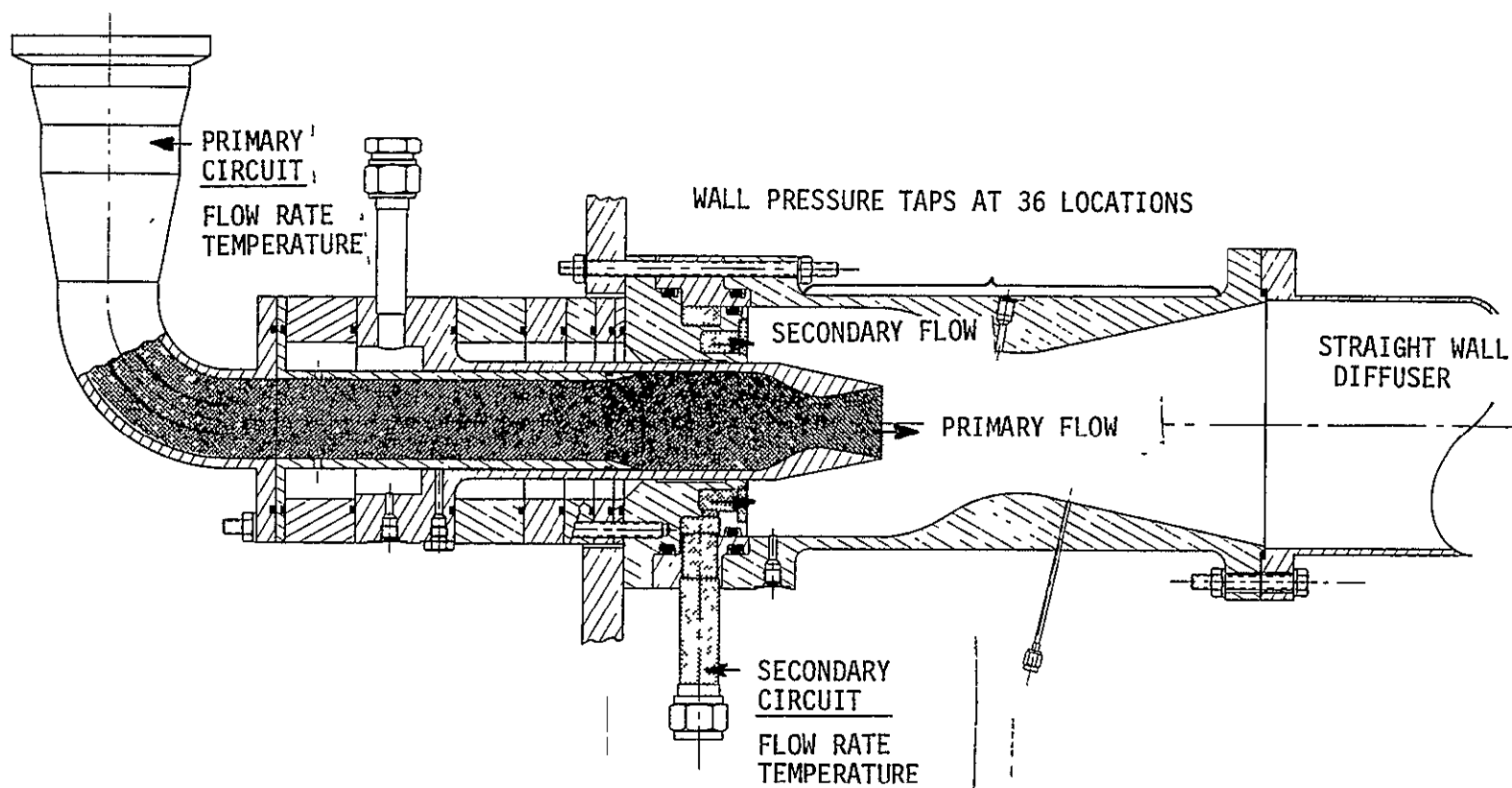


Figure 6. Dual Throat Assembly as Tested

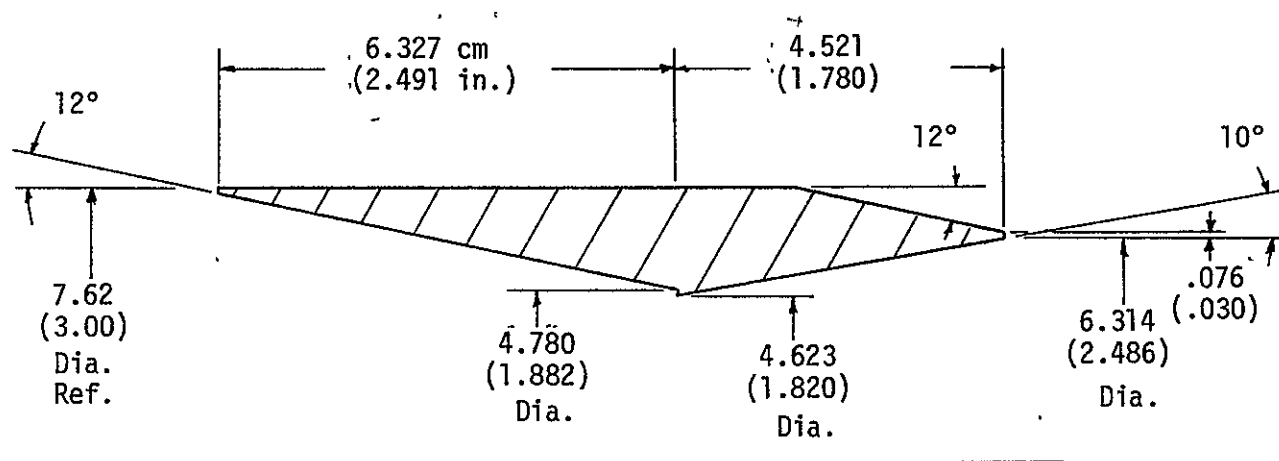


Figure 7. Dual Throat Nozzle Extension

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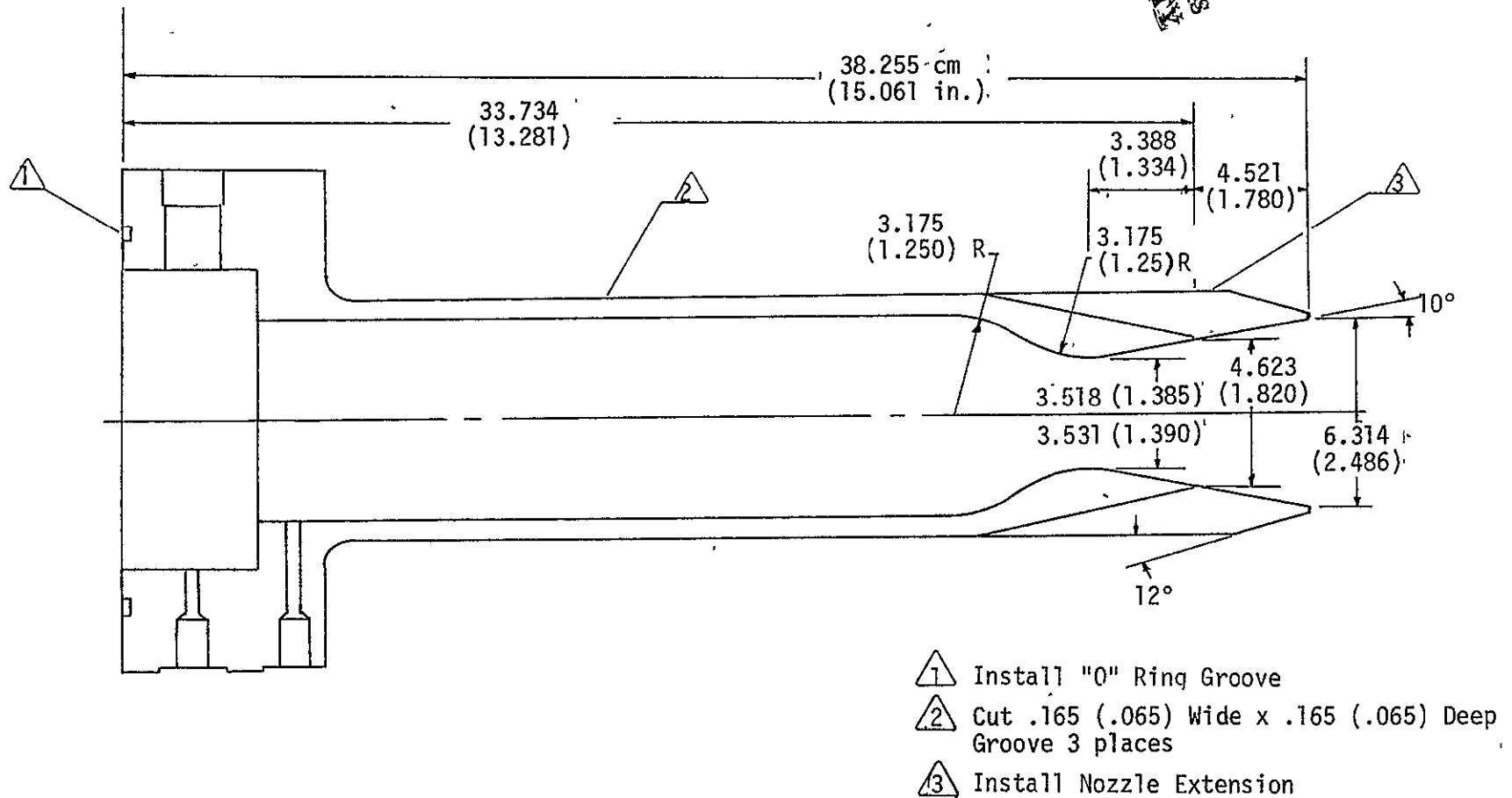


Figure 8. Primary Hardware Modifications

III, B, Experimental Apparatus (cont.)

2. Test Facility

Figure 9 shows a schematic diagram of the test set up. The major components were the gaseous nitrogen storage tank, tank safety valve, pressure regulators, pebble bed heat storage device, electric heater, flow measurement nozzles, thrust chamber valves, and the test engine. Two flow circuits were provided, a low flow circuit with a run line fabricated from 3.81 cm (1.5 in.) stainless steel tubing, and a high flow circuit which used 5.08 cm (two-inch) stainless pipe upstream of the flow measurement nozzle and 7.62 cm (three-inch) pipe downstream. The high flow circuit contained a pebble bed heat storage device which heated the nitrogen to prevent condensation shocks from occurring in the nozzle. The pebble bed could maintain the temperature at the inlet to the engine at $355.6 \pm 18^\circ\text{K}$ ($180 \pm 10^\circ\text{F}$) for 15 sec, at a flow rate of 4.54 Kg/s (10 lb/sec). Figures 10 and 11 show front and rear views of the dual throat assembly on the test stand.

3. Instrumentation

The engine was instrumented for measuring the flow in both primary and secondary circuits, gas temperature and pressure in both circuits, primary and secondary chamber pressure, and 36 wall pressure measurements connected to two scanivalves. Each scanivalve consisted of a pressure transducer close coupled to a solenoid operated pneumatic switch. The pneumatic switch had 48 input ports and the system was capable of stepping from one port to the next at rates up to 30 steps per second. For this investigation two scanivalves were used; one for low pressure and one for high pressure. The pressure taps were connected to the scanivalves with plastic tubing. Each pressure tap was connected to two scanivalve ports so that two separate readings could be obtained each time the scanivalve made a complete cycle (see Figure 12). An ambient pressure and a reference pressure were also connected to each scanivalve. Table I lists the instrumentation.

The heart of the instrumentation system was a mini-computer. This computer controlled the test, and gathered and processed the test data. Two special computer programs were written for the dual throat testing. "APLME" was a special data reduction program and "ASCAN" was a modified sequencing program. ASCAN performed the normal sequencing functions such as operating the thrust chamber valves and turning on the oscillograph at operator selected times, checking for malfunctions, and acquiring data and converting it to engineering units based on calibration information; but had the additional capability of operating the scanivalves. The number of pressure measurements connected to each scanivalve, number of data records obtained, and the time between completion of a scanivalve step and reading the pressures were input variables to the program.

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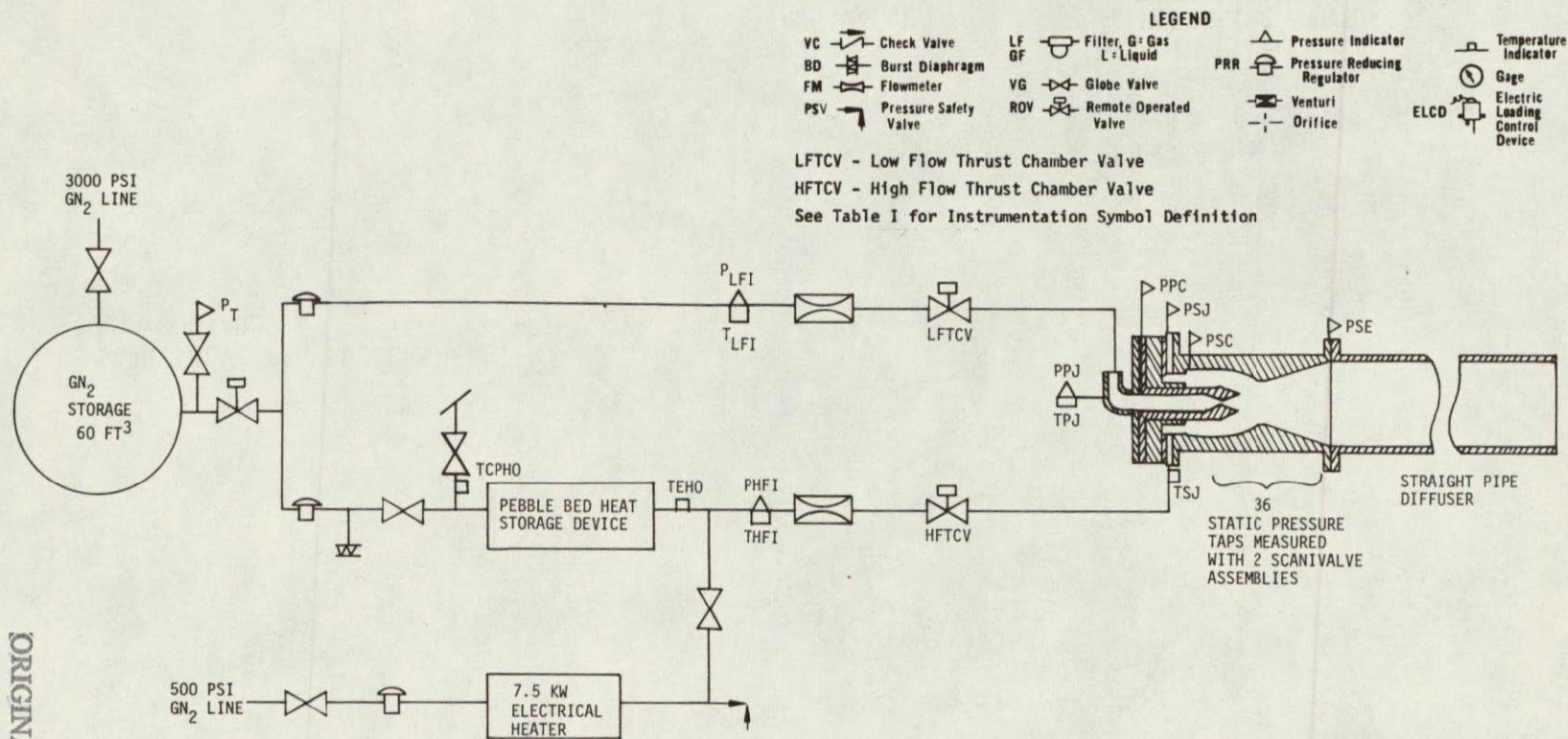


Figure 9. Test Stand Schematic

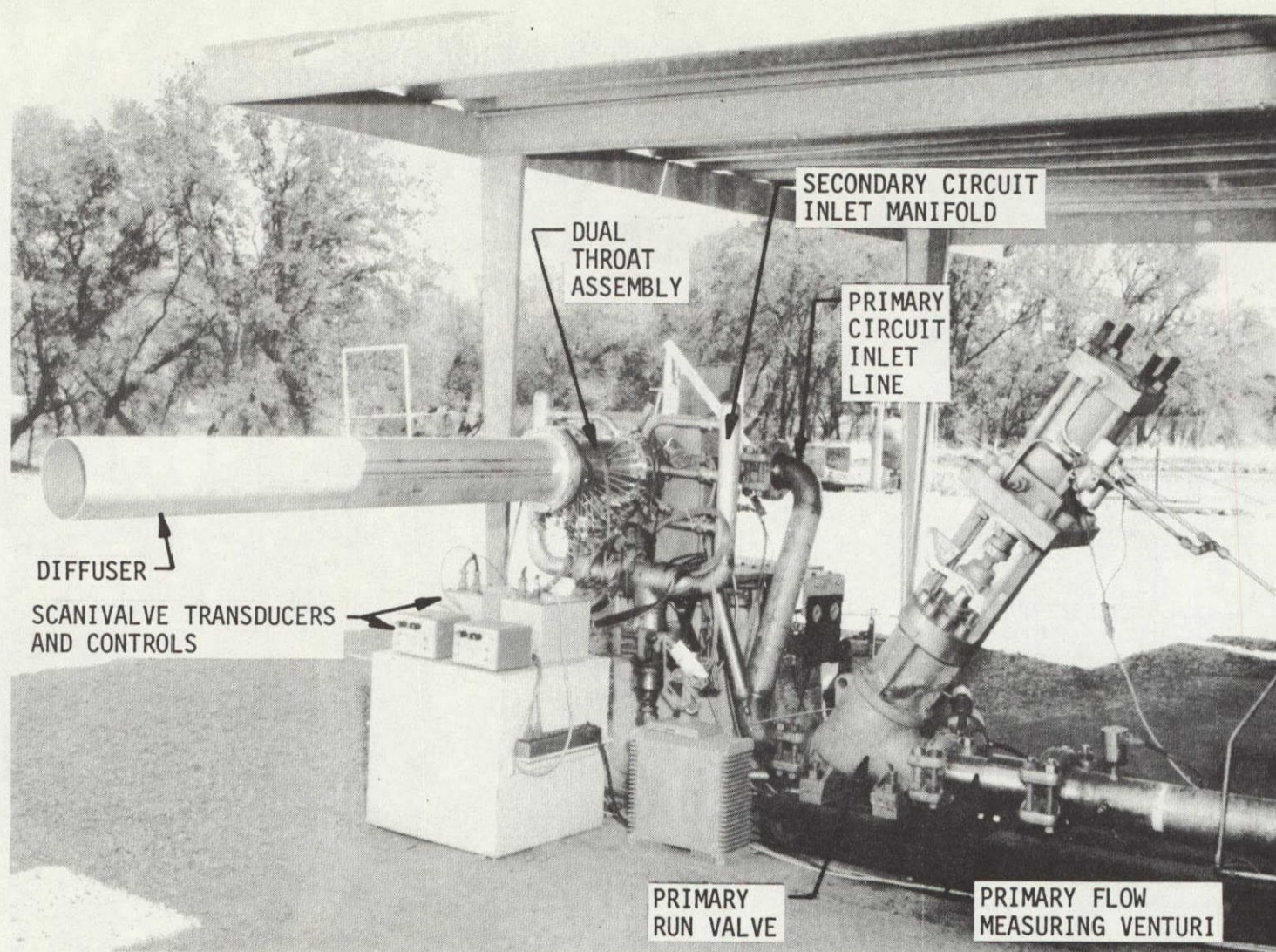


Figure 10. Dual Throat Cold Flow Test Stand Front View

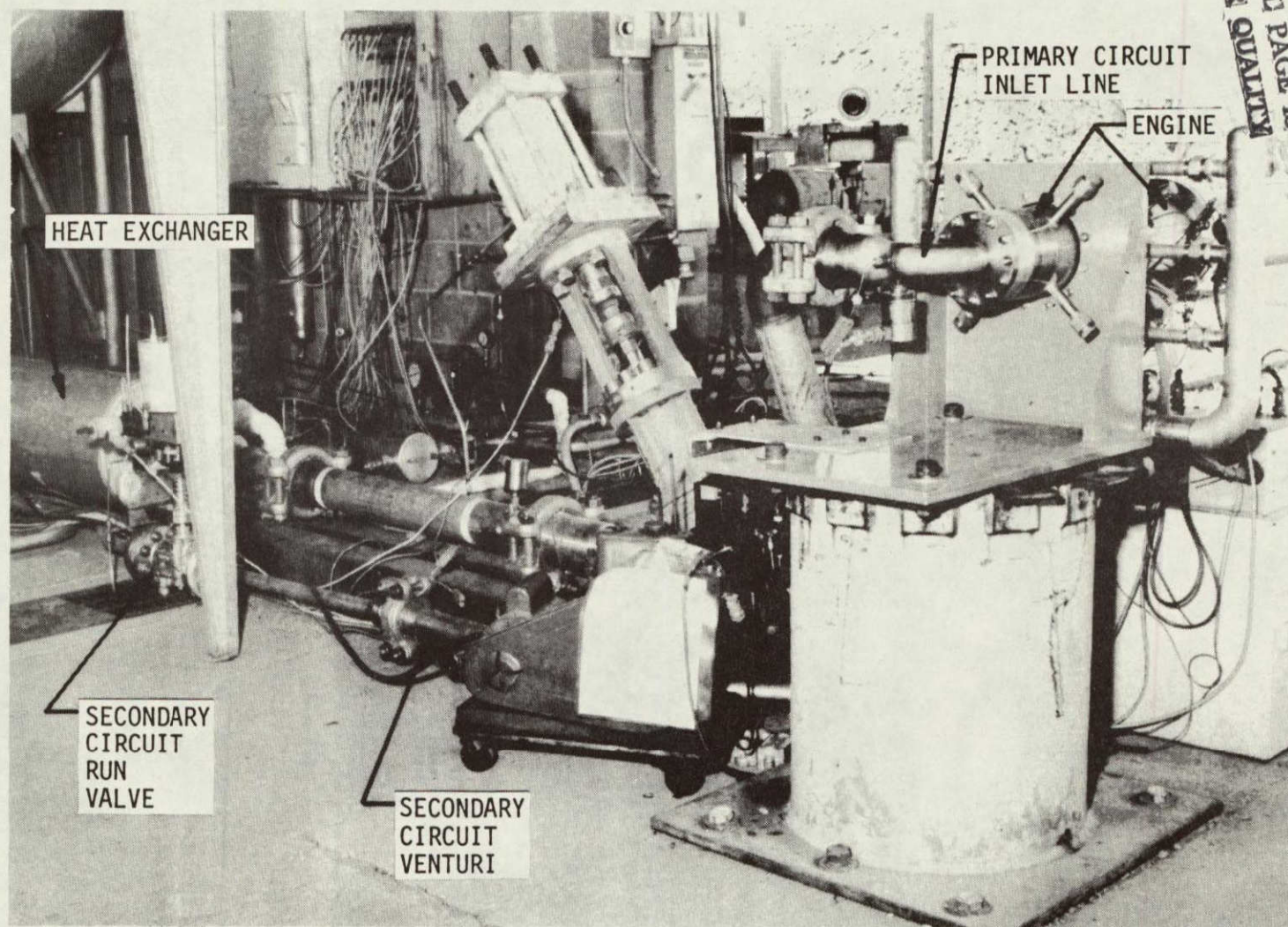


Figure 11. Rear View of Cold Flow Test Stand

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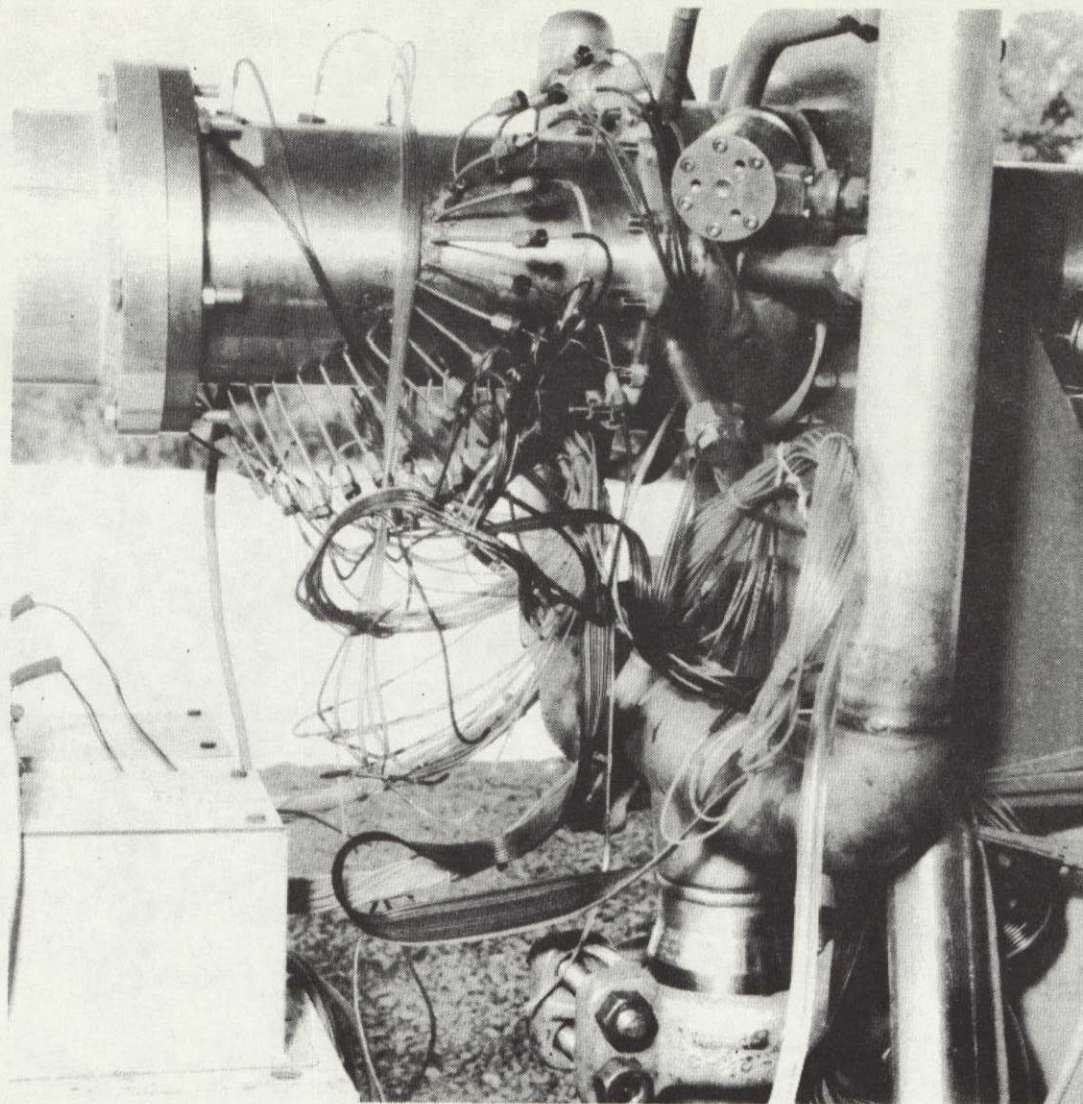


Figure 12. Secondary Chamber Pressure Instrumentation

III, Dual-Throat Cold Flow Test Data (cont.)

C. TEST PROCEDURE

Three test series (40 tests) were conducted under ALRC IR&D funding. The configurations tested are shown in Figure 2. An additional 29 Mode II (Series IA) tests were completed on this contract. The hardware configurations for these tests are illustrated in Figure 3. Cold flow test history is summarized in Figure 13.

For Series I and IA (Plume Attachment Tests) the high flow circuit was connected to the primary chamber. The primary flow rate was 4.54 kg/s (10 lbm/sec) for the entire series, and the bleed flow was varied from zero to 0.681 kg/s (1.5 lbm/sec) for various primary nozzle positions.

For Test Series II and III the high flow circuit was connected to the secondary chamber. A secondary flow of 6.36 kg/s (14 lbm/sec) was used for Test Series II (Secondary Blockage Tests). There was no flow in the primary circuit for this series and five nozzle positions were used.

Test Series III (Parallel Flow Tests) was conducted at a single nozzle position with a total flow of 6.81 kg/s (15 lbm/sec) and primary flow fractions of 12 to 29%. For this series both primary and secondary flow leads were investigated.

All test series were conducted according to the following basic procedure. First the electric heater was turned on and heated nitrogen was allowed to flow through the pebble bed heat storage device. Approximately one hour was required to heat the pebble bed. Two seven second tests could be run on each heating. While the pebble bed was heating, the primary nozzle was adjusted to the proper position as specified in the test plan, and final instrumentation calibrations were obtained. When the pebble bed had reached 367°K (200°F) the electric heater was turned off and the preheat valves closed. The tank safety valve was then opened, and the pressure regulators set to give the required flow in each circuit. The arm and fire switches were then depressed which turned control over to the mini-computer. One half second after the fire switch was pushed, the computer turned on the oscillograph and the digital data system. Another half second later the thrust chamber valves were opened. (For Test Series III the valves were opened sequentially two seconds apart, for Series II only the high flow valve was opened, and for Series I the valves were opened simultaneously.) The pressures were allowed to stabilize for two seconds after opening the valves, before the computer called for the first data record. For each data record all important parameters were recorded including one pressure measurement for each scanivalve. After each record was obtained the scanivalve was advanced one step. When the computer received a signal from the scanivalve indicating the step had been completed and the scanivalve was in

● ALRC IR&D TESTS	- 40 TESTS
MODE II - PLUME ATTACHMENT TESTS	
4 NOZZLE POSITIONS	} - 21 TESTS
0-10% BLEED FLOWS	
MODE I - SECONDARY BLOCKAGE	
5 NOZZLE POSITIONS	- 5 TESTS
MODE I - PARALLEL BURN	- 14 TESTS
● NASA-MSFC CONTRACTUAL TESTS	
MODE II - PLUME ATTACHMENT TESTS	- 29 TESTS
VERIFICATION OF 2 NOZZLE POSITIONS	- 9 TESTS
HARDWARE CONFIGURATION VARIATIONS	
2 PRIMARY NOZZLE AREA RATIOS	} - 20 TESTS
2 SECONDARY NOZZLE AREA RATIOS	

Figure 13. Dual Throat Cold Flow Test History

III, C, Test Procedure (cont.)

position for the next pressure reading, the computer waited 80 milliseconds for the pressure reading to stabilize before calling for another data record. (Prior to starting the test program, the scanivalve was connected to two known pressure sources and stepped from one to the other while the output signal was recorded on an oscillograph. The results indicated a delay of 80 milliseconds was sufficient to insure accurate pressure readings.) After stepping through the scanivalve channels, the computer terminated the test by closing the thrust chamber valves. The computer then printed a summary of the test data and transferred the average value of each parameter to the data reduction program "APLME". APLME first corrected all the scanivalve measurements based on the reference pressure readings, and tabulated the results according to axial distance from the inlet of the secondary chamber as shown in Table II. (This was effectively calibrating the scanivalve transducers during each test run, and eliminated any errors due to instrumentation drift.) APLME next calculated the flow rate for each circuit and for Test Series I also computed the thrust coefficient and specific impulse from the pressure measurements. A sample printout is given in Table III.

D. DATA ANALYSIS

1. Series I and IA - Plume Attachment Test Results

The wall pressure integration technique shown in Figure 14 for calculating thrust is estimated to have an accuracy of ± 0.5 percent. For this reason and in an effort to reduce program scope, no direct thrust measurements were made. As indicated in the figure, the total engine thrust is the summation of the primary nozzle thrust (which was obtained using the thrust coefficient equation) and the integrated wall pressure thrust. The engine specific impulse is the total engine thrust divided by the total gas flow rate (primary flow plus bleed flow).

A procedure was developed to isolate the losses associated with the dual throat engine concept, i.e., the plume attachment losses and bleed flow losses. The test one-dimensional engine specific impulse (I_{spODE}) was calculated as the mass average of the primary and bleed flow circuits I_{spODE} values. The I_{spODE} for each circuit was based upon a calculation for expanding nitrogen gas to an area ratio of 20 to 1 using the respective circuit gas chamber temperatures. (The dual throat hardware Mode II area ratio was 20:1.) The percent I_{sp} loss due to the dual throat concept was then taken to be the I_{spTDK} for a conventional nozzle with the same conical half angle as the dual throat hardware ($\theta = 12.5^\circ$) minus the measured test I_{sp} divided by the I_{spODE} minus the percent I_{sp} loss due to the viscous drag within the boundary layer of the nozzle at the cold flow test conditions. This loss was calculated using the TBL chart program and was found to be less than 0.1% and, therefore, was ignored in this analysis. This cal-

TABLE II

CORRECTED WALL PRESSURES

DUAL THROAT HARDWARE N2 COOLD FLOW TEST PROGRAM

TEST NO. CFR6-727-109

TEST DATE 02-08-78 12.30.43

BAROMETRIC PRESSURE 14.483

REFERENCE PRESSURE .01934 .01934

SECTION 3 -- SECONDARY CHAMBER WALL PRESSURE FROM SCAN/VALVE INPUT

1ST SCAN													
X (IN)	R (IN)	TIME (SEC)	PW1 (PSIA)	PCH (PSIA)	PW1/PCH	TIME (SEC)	PW2 (PSIA)	PCH (PSIA)	PW2/PCH	TIME (SEC)	PW3 (PSIA)	PCH (PSIA)	PW3/PCH
2.940	2.925	2.089	11.650	323.815	.0360	3.150	11.669	324.846	.0359	.000	.000	.000	.0000
4.437	2.872	2.239	11.700	324.053	.0361	.000	.000	.000	.0000	.000	.000	.000	.0000
5.190	2.621	2.361	11.726	324.370	.0361	3.300	11.759	325.375	.0361	.000	.000	.000	.0000
5.939	2.197	2.497	11.709	324.688	.0361	3.422	11.754	325.719	.0361	3.830	11.739	326.406	.0360
6.429	1.971	2.089	11.596	323.815	.0358	.000	.000	.000	.0000	.000	.000	.000	.0000
6.940	1.837	2.239	11.413	324.053	.0352	3.681	11.519	326.433	.0353	4.361	11.407	327.754	.0348
7.189	1.806	2.361	11.305	324.370	.0349	.000	.000	.000	.0000	.000	.000	.000	.0000
7.440	1.795	2.497	11.301	324.688	.0348	3.830	11.372	326.406	.0348	4.483	10.903	328.098	.0332
7.691	1.806	2.620	10.429	324.688	.0321	.000	.000	.000	.0000	.000	.000	.000	.0000
7.940	1.837	2.769	8.970	324.529	.0276	3.953	8.790	326.406	.0269	4.619	8.791	328.072	.0268
8.441	1.944	2.892	6.272	324.820	.0193	4.089	6.263	327.120	.0191	.000	.000	.000	.0000
8.940	2.055	3.028	5.073	324.688	.0156	.000	.000	.000	.0000	.000	.000	.000	.0000
9.440	2.166	3.150	4.380	324.846	.0135	.000	.000	.000	.0000	.000	.000	.000	.0000
9.939	2.276	3.300	3.998	325.375	.0123	4.211	3.969	327.437	.0121	4.741	3.832	328.548	.0117
10.438	2.387	3.422	3.365	325.719	.0103	.000	.000	.000	.0000	.000	.000	.000	.0000
10.938	2.497	3.558	2.836	326.300	.0087	.000	.000	.000	.0000	.000	.000	.000	.0000
11.437	2.608	3.620	2.466	324.688	.0076	.000	.000	.000	.0000	.000	.000	.000	.0000
11.937	2.719	2.769	2.075	324.529	.0064	3.558	2.142	326.300	.0066	3.953	2.106	326.406	.0065
12.436	2.830	.000	.000	.000	.0000	3.681	1.781	326.433	.0055	.000	.000	.000	.0000
12.927	2.939	2.892	1.604	324.820	.0049	.000	.000	.000	.0000	.000	.000	.000	.0000
13.508	3.067	3.028	1.315	324.688	.0041	.000	.000	.000	.0000	.000	.000	.000	.0000
2ND SCAN													
2.940	2.925	4.877	11.776	328.785	.0358	5.938	11.820	329.764	.0358	.000	.000	.000	.0000
4.437	2.872	5.013	11.830	328.891	.0360	.000	.000	.000	.0000	.000	.000	.000	.0000
5.190	2.621	5.149	11.837	329.024	.0360	6.061	11.902	329.975	.0361	.000	.000	.000	.0000
5.939	2.197	5.272	11.833	329.261	.0359	6.197	11.887	330.689	.0359	6.591	11.886	331.376	.0359
6.429	1.971	4.877	11.715	328.785	.0356	.000	.000	.000	.0000	.000	.000	.000	.0000
6.940	1.837	5.013	11.551	328.891	.0351	6.469	11.656	330.557	.0353	7.121	11.535	331.800	.0348
7.189	1.806	5.149	11.442	329.024	.0348	.000	.000	.000	.0000	.000	.000	.000	.0000
7.440	1.795	5.272	11.350	329.261	.0345	6.591	11.580	331.376	.0349	7.257	11.050	331.905	.0333
7.691	1.806	5.408	10.627	329.288	.0323	.000	.000	.000	.0000	.000	.000	.000	.0000
7.940	1.837	5.530	9.122	329.420	.0277	6.727	8.968	331.958	.0270	7.380	8.933	332.355	.0269
8.441	1.944	5.680	6.362	329.790	.0193	6.849	6.359	331.644	.0192	.000	.000	.000	.0000
8.940	2.055	5.802	5.156	329.473	.0156	.000	.000	.000	.0000	.000	.000	.000	.0000
9.440	2.166	5.938	4.453	329.764	.0135	.000	.000	.000	.0000	.000	.000	.000	.0000
9.939	2.276	6.061	4.082	329.975	.0124	6.985	4.016	331.376	.0121	7.516	3.869	332.619	.0116
10.438	2.387	6.197	3.411	330.689	.0103	.000	.000	.000	.0000	.000	.000	.000	.0000
10.938	2.497	6.319	2.870	330.716	.0087	.000	.000	.000	.0000	.000	.000	.000	.0000
11.437	2.608	5.408	2.494	329.288	.0076	.000	.000	.000	.0000	.000	.000	.000	.0000
11.937	2.719	5.530	2.103	329.420	.0064	6.319	2.168	330.716	.0066	6.727	2.147	331.958	.0065
12.436	2.830	.000	.000	.000	.0000	6.469	1.807	330.557	.0055	.000	.000	.000	.0000
12.927	2.939	5.680	1.604	329.790	.0049	.000	.000	.000	.0000	.000	.000	.000	.0000
13.508	3.067	5.802	1.291	329.473	.0039	.000	.000	.000	.0000	.000	.000	.000	.0000

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TABLE III

FLOW AND PERFORMANCE CALCULATION (1 of 3)

DUAL THROAT GN2 COLD FLOW TEST PROGRAM

TEST NO. CFB6-727-109

TEST DATE 02-08-78 12.30.43

PROGRAM INPUT

TYPE # 1

TYPE 1 = MODE 2 PLUME ATTACHMENT TEST

TYPE 2 = MODE 1 SERIES BURN TEST

TYPE 3 = MODE 1 PARALLEL BURN TEST

LERTP = 2.883

PRIMARY FLOW RATE (LBM/SEC)	10.004
SECONDARY FLOW RATE (LBM/SEC)	.590
PRIMARY STATIC CHAMBER PRESSURE (PSIA)	328.406
PRIMARY GAS TEMPERATURE (R)	663.629
SECONDARY GAS TEMPERATURE (R)	511.735

AVERAGE OF WALL PRESSURES (PSIA)

#	X (INCHES)	R (INCHES)	PRESSURE
1	2.940	2.925	11.786
2	4.437	2.872	11.835
3	5.190	2.621	11.850
4	5.939	2.197	11.815
5	6.429	1.971	11.731
6	6.940	1.837	11.519
7	7.189	1.806	11.433
8	7.440	1.795	11.252
9	7.691	1.806	10.574
10	7.940	1.837	8.919
11	8.441	1.944	6.315
12	8.940	2.055	5.135
13	9.440	2.166	4.431
14	9.939	2.276	3.951
15	10.438	2.387	3.390
16	10.938	2.497	2.852
17	11.437	2.608	2.490
18	11.937	2.719	2.125
19	12.436	2.830	1.794
20	12.927	2.939	1.609
21	13.508	3.067	1.309

AVERAGE PRIMARY NOZZLE OUTER WALL PRESSURE
PNBAR = 11.817 PSIA

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TABLE III

FLOW AND PERFORMANCE CALCULATION (2 of 3)

DUAL THROAT GN2 COLD FLOW TEST PROGRAM

TEST NO. CFB6-727-109

TEST DATE 02-08-78 12.30.43

PROGRAM INPUT

TYPE = 1

TYPE 1 = MODE 2 PLUME ATTACHMENT TEST

TYPE 2 = MODE 1 SERIES BURN TEST

TYPE 3 = MODE 1 PARALLEL BURN TEST

LERTP = 2.883

PRESSURE CALCULATIONS

	STATIC	STAGNATION
PRIMARY	328,406	339,320
SECONDARY	11,786	11,809

POS/POP = .035

FLOW CALCULATIONS

	PRIMARY	SECONDARY
FLOW RATE (LB/SEC)	10.000	5.590
MAX. IDEAL FLOW	10.146	2.767
PERCENT OF MAX.	98.598	21.325
NORMALIZED FLOW	.759	1.131
FLOW COEFFICIENT	.986	.213

% BLEED FLOW = 5.570

TABLE III

FLOW AND PERFORMANCE CALCULATION (3 of 3)

DUAL THROAT GN2 COLD FLOW TEST PROGRAM

TEST NO. CFB6-727-100

TEST DATE 02-08-78 12.30.43

PROGRAM INPUT

TYPE = 1

TYPE 1 = MODE 2 PLUME ATTACHMENT TEST

TYPE 2 = MODE 1 SERIES BURN TEST

TYPE 3 = MODE 1 PARALLEL BURN TEST

LERTP = 2.883

THRUST CALCULATIONS

PRIMARY EXPANSION

PRIMARY NOZZLE THRUST (LBF) = 702.732

NOZZLE WALL PRESSURE INTEGRATION

**THRUST GENERATED ALONG PATH FROM PRIMARY NOZZLE EXIT TO SECONDARY CHAMBER.

NORMAL PROJECTED AREA = 24.426 IN²

BACK PRESSURE = 11.824 PSIA

THRUST = 286.805 LBFs

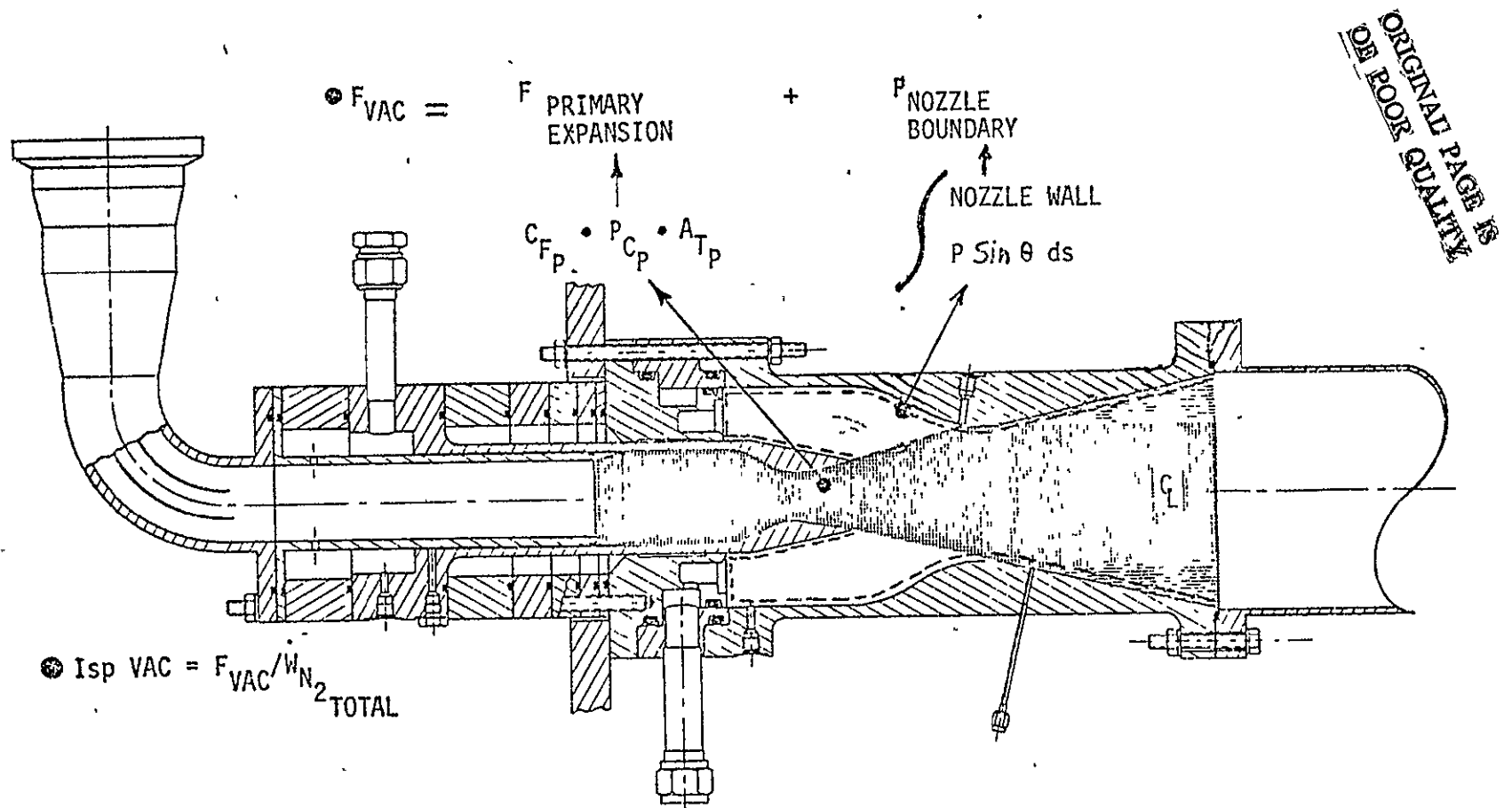
**THRUST GENERATED ALONG PATH FROM SECONDARY CHAMBER TO SECONDARY NOZZLE EXIT.

X (IN)	R (IN)	S (IN ²)	THET (DEG)	PW (PSIA)	DFSE (LBF)	DFSES (LBF)
2.940	2.925	-.965	0.000	11.786	-11.400	277.405
4.437	2.872	-4.331	-10.800	11.835	-81.295	226.110
5.190	2.621	-6.418	-26.000	11.850	-75.938	150.172
5.939	2.197	-2.959	-30.000	11.815	-34.840	115.332
6.429	1.971	-1.603	-19.700	11.731	-18.636	96.697
6.940	1.837	-.355	-9.600	11.519	-4.072	92.625
7.189	1.806	-.124	-4.800	11.433	-1.411	91.214
7.440	1.795	.124	.000	11.252	1.358	92.572
7.691	1.806	.355	4.800	10.574	3.458	96.030
7.940	1.837	1.271	9.600	8.919	9.681	105.711
8.441	1.944	1.395	12.500	6.315	7.984	113.694
8.940	2.055	1.472	12.500	5.135	7.040	120.735
9.440	2.166	1.535	12.500	4.431	6.433	127.168
9.939	2.276	1.626	12.500	3.951	5.968	133.136
10.438	2.387	1.688	12.500	3.390	5.268	138.404
10.938	2.497	1.780	12.500	2.852	4.755	143.159
11.437	2.608	1.858	12.500	2.490	4.286	147.445
11.937	2.719	1.935	12.500	2.125	3.792	151.237
12.436	2.830	1.975	12.500	1.794	3.361	154.598
12.927	2.939	2.415	12.500	1.609	3.524	158.122
13.508	3.067	.834	12.500	1.309	1.051	159.173

PERFORMANCE SUMMARY

METHOD	THRUST (LBF)	ISP (SEC)	CF	ODE (SEC)	% LOSS ISP
1	861.905	81.356	1.725	82.737	.969
TOTAL GAS FLOW RATE =			10.594	LBM/SEC	

● PRESSURE INTEGRATION TECHNIQUE, THRUST MEASUREMENT ACCURACY = $\pm 0.5\%$



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Figure 14. Dual Throat Cold Flow Thrust Calculation

III, D, Data Analysis (cont.)

ulation, as described in Figure 15, normalizes the loss to make it independent of the nozzle contour.

Plume attachment test data for Series I (IR&D Program) are summarized in Table IV.

The wall pressure profiles for each test, plotted as a function of axial distance from the injector face for various amounts of bleed flows, are shown in Figures 16 to 19. Superimposed on each pressure plot is a profile of the two nozzles showing their relative positions. For the cases of zero or near zero bleed flows the presence of the plume attachment shock is evident by the sudden increase in the wall pressure. As bleed flow is added and the back pressure increases, the plume is turned and the attachment deflection angle is reduced to the point where the shock is eliminated.

The corresponding performance data are also shown plotted with each pressure profile. These dual throat concept performance losses are plotted as a function of bleed flow percentage. In each plot, the performance losses reach a minimum for bleed flows of 4-8 percent. The percent bleed flow corresponding to the minimum performance loss also corresponds to the minimum bleed flow necessary to eliminate the wall pressure spike (plume attachment shock). This trend holds for all four positions tested during the IR&D program. A summary of the performance losses of the four positions is given in Figure 20. A cross plot of these data is shown in Figure 21. These plots indicate that the lowest loss of about 0.5% occurs at an L_e/R_{tp} of about 3.0, with approximately 4.0 percent bleed flow. The rapid increase in the performance loss for zero bleed flow for the larger nozzle separation distances is a result of the plume attaching upstream of the secondary nozzle throat. As this occurs the plume attachment deflection angle is greater and the resultant shock losses are increased. The optimum nozzle spacing seems to be obtained when the plume is made to attach tangent to the wall at the secondary throat plane while using the smallest amount of bleed flow.

A summary of the contractual testing (Series IA) is given in Table V. The results of the first 9 verification tests are plotted on Figures 18-20 and compared to the IR&D (Series I) results. Agreement is good between the two data sets. The apparent bias of 0.5 to 1.0% is believed to be an instrumentation bias as the pressure transducer used to measure the upstream pressure of the metering venturi in the high flow circuit was decreased from $20.68 \times 10^6 \text{ N/m}^2$ (3000 psi) to $13.78 \times 10^6 \text{ N/m}^2$ (2000 psi). The resultant flow measurement uncertainty remained the same at $\pm 1\%$.

$$I_{spODE} = \text{MASS AVERAGE } (I_{spODEPRI} + I_{spODESEC})$$

$$\left. \begin{aligned} I_{spODEPRI} &= f(T_p) \\ I_{spODESEC} &= f(T_s) \end{aligned} \right\} \begin{array}{l} \text{NITROGEN} \\ \epsilon = 20:1 \end{array}$$

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• Dual Throat Concept Loss Calculation

$$\% I_{sp} \text{ Loss} = \frac{I_{spTDK}^* - I_{spTest}}{I_{spODE}} - \% I_{spBL}^*$$

where: $\frac{I_{spTDK}}{I_{spODE}} = 99.3 \% \text{ for a } 12.5^\circ \text{ conical nozzle}$

Figure 15. Dual Throat Cold Flow Performance Evaluation

*Percent Isp loss due to boundary layer viscous drag for the conventional 12.5 degree conical nozzle (loss was 0 to 0.1% for the cold flow tests).

TABLE IV. - DUAL THROAT COLD FLOW SERIES I TEST SUMMARY
ALRC IR&D, MODE II TESTS

TEST	TEMPERATURE °K		CHAMBER STAGNATION PRESSURE, N/cm ²		% BLEED	L / R e tp	F _{Vac} (KN)	I _{sp} TEST SEC.	I _{sp} CODE SEC.	%I _{sp} LOSS	e _p	e _s
	PRIMARY	SECOND	PRIMARY	SECOND								
101	361	302	195	1.7	0.0	1.067	3.02	80.7	82.5	1.39	1.666	2.979
102	354	282	196	3.3	4.8	1.067	3.17	79.8	81.2	1.04		
103	366	286	203	5.4	7.8	1.067	3.38	80.1	82.2	1.90		
104	361	277	210	9.8	14.2	1.067	3.64	77.9	80.9	3.10		
106	356	303	203	3.6	0.0	2.883	3.13	79.8	81.9	1.90		
108	363	291	204	6.6	4.2	2.883	3.31	81.0	82.3	0.82		
109	356	286	208	8.6	8.2	2.883	3.48	79.7	81.2	1.16		
110	355	298	201	5.1	2.0	2.883	3.20	80.4	81.6	0.76		
111	351	299	207	4.8	1.4	2.883	3.26	79.9	81.2	.90		
112	360	307	199	4.7	0.0	5.045	2.89	75.9	82.3	7.12		
113	358	297	203	6.7	4.0	5.045	3.19	78.8	81.8	3.01		
114	362	288	203	8.1	7.2	5.045	3.31	79.3	82.0	2.57		
115	359	282	205	9.4	10.3	5.045	3.43	78.1	81.3	3.20		
116	361	294	203	3.9	3.1	1.067	3.22	79.9	82.2	2.15		
117	356	324	206	3.1	1.3	1.067	3.23	80.1	81.8	1.35		
118	364	299	207	4.0	2.4	1.067	3.28	80.9	82.6	1.38		
119	358	304	209	1.9	0.0	1.067	3.22	80.2	82.1	1.67		
120	360	300	203	1.2	0.0	0.0	3.12	79.8	82.4	2.46		
121	355	285	206	3.7	3.5	0.0	3.26	79.2	81.5	2.10		
122	362	279	199	5.0	7.5	0.0	3.22	78.4	81.8	3.44		
123	357	281	205	4.5	5.7	0.0	3.29	78.7	81.5	2.74		

TABLE IV (cont.)

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TEST	TEMPERATURE, °R		CHAMBER STAGNATION PRESSURE, PSIA		% BLEED	L_e/R_{tp}	F_{vac} LBF	I_{sp} TEST		I_{sp}^{ODE}	% I_{sp} LOSS	t_p	t_s
	PRIMARY	SECOND	PRIMARY	SECOND				SEC	SEC				
101	650	543	283	2.5	0.0	1.067	679	80.7	82.5	1.39		1.666	2.979
102	637	508	284	4.8	4.8	1.067	713	79.8	81.2	1.04			
103	658	514	295	7.8	7.8	1.067	759	80.1	82.2	1.90			
104	649	498	304	14.2	14.2	1.067	819	77.9	80.9	3.10			
105	641	545	295	5.2	0.0	2.883	703	79.8	81.9	1.90			
108	653	523	296	9.5	4.2	2.883	743	81.0	82.3	0.82			
109	641	515	301	12.5	8.2	2.883	782	79.7	81.2	1.16			
110	639	537	292	7.4	2.0	2.883	720	80.4	81.6	0.76			
111	631	538	300	6.9	1.4	2.883	734	79.9	81.2	.90			
112	648	552	289	6.8	0.0	5.045	650	75.9	82.3	7.12			
113	644	535	294	9.7	4.0	5.045	718	78.8	81.8	3.01			
114	652	519	294	11.8	7.2	5.045	744	79.3	82.0	2.57			
115	647	507	298	13.7	10.3	5.045	771	78.1	81.3	3.20			
116	650	530	295	5.7	3.1	1.067	724	79.9	82.2	2.15			
117	641	534	299	4.5	1.3	1.067	726	80.1	81.8	1.35			
118	655	538	300	5.8	2.4	1.067	737	80.9	82.6	1.38			
119	644	548	303	2.8	0.0	1.067	724	80.2	82.1	1.67			
120	648	540	295	1.7	0.0	0.0	701	79.8	82.4	2.46			
121	639	513	299	5.3	3.5	0.0	734	79.2	81.5	2.10			
122	651	502	288	7.3	7.5	0.0	725	78.4	81.8	3.44			
123	643	506	297	6.5	5.7	0.0	739	78.7	81.5	2.74			

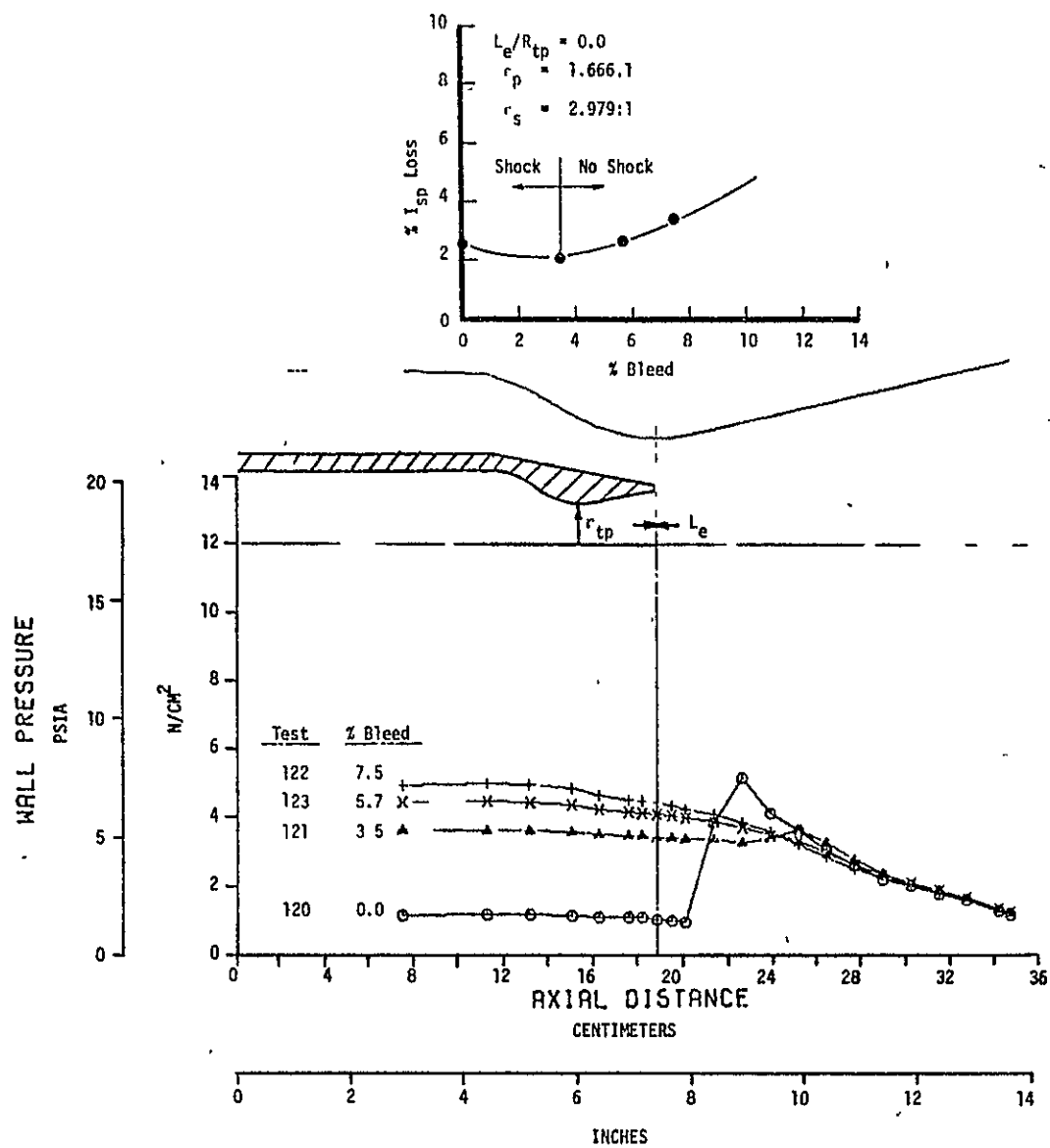


Figure 16. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{TP} = 0.0$)

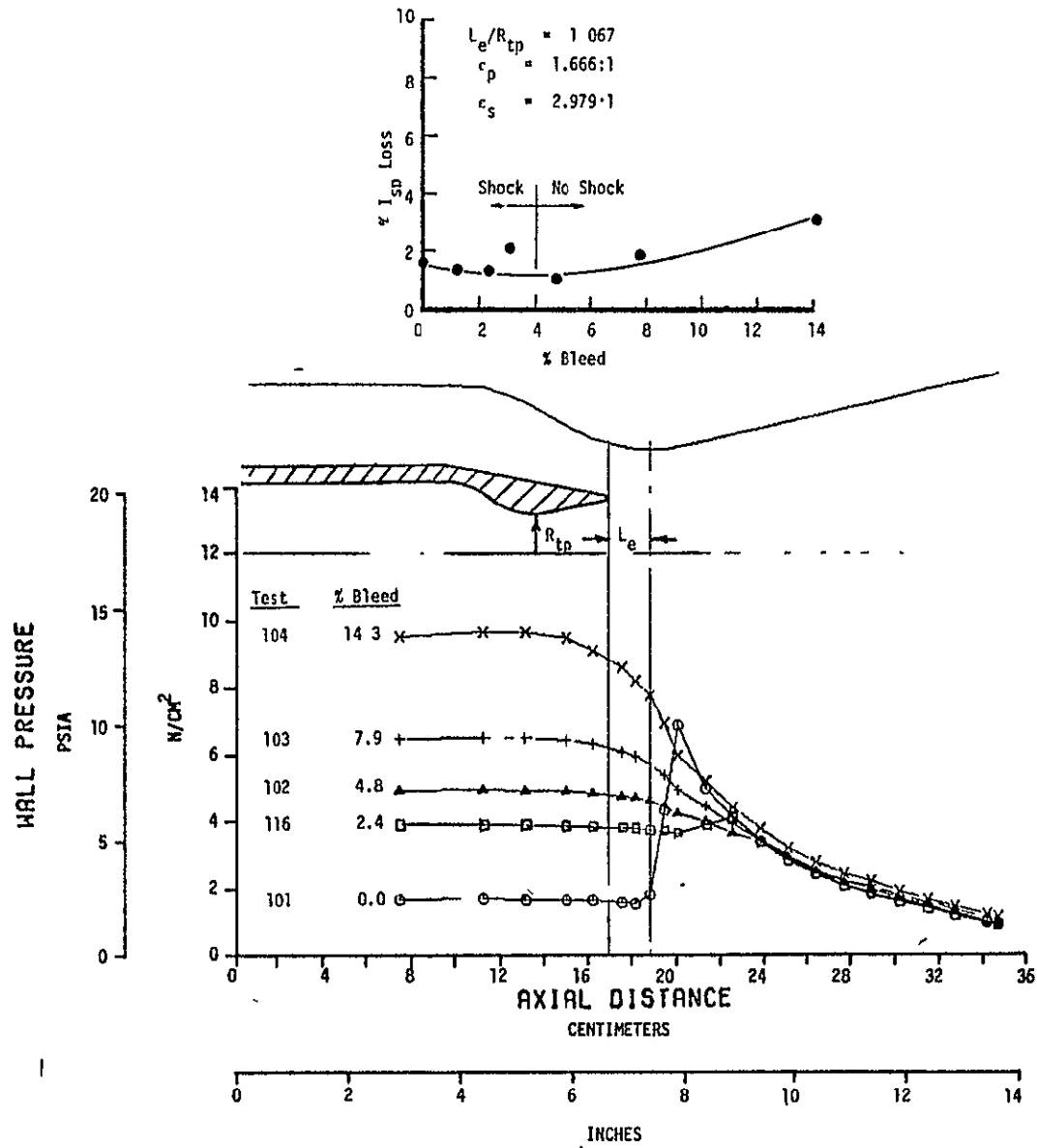


Figure 17. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{tp} = 1.067$)

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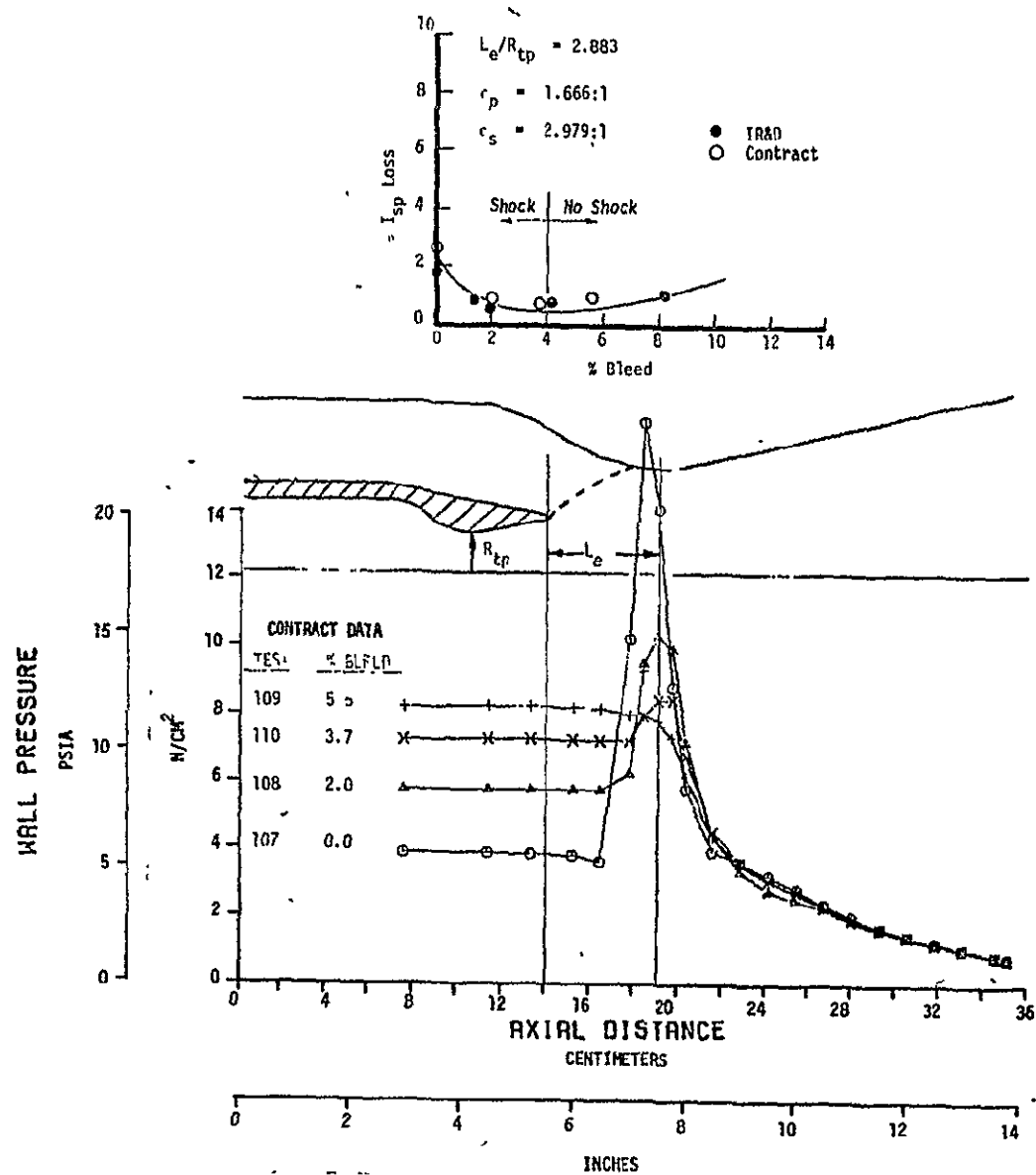


Figure 18. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{tp} = 2.883$)

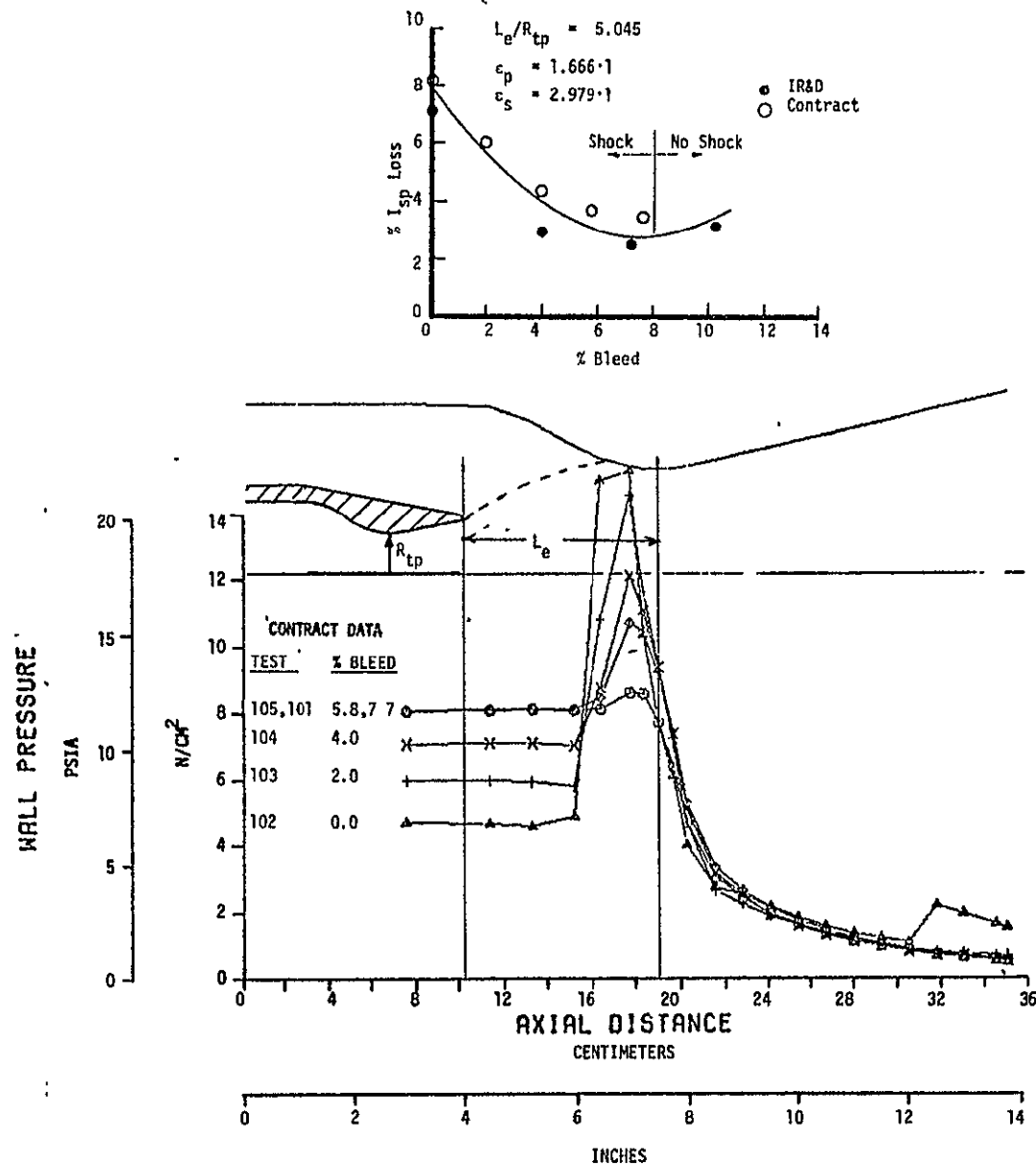


Figure 19. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{TP} = 5.045$)

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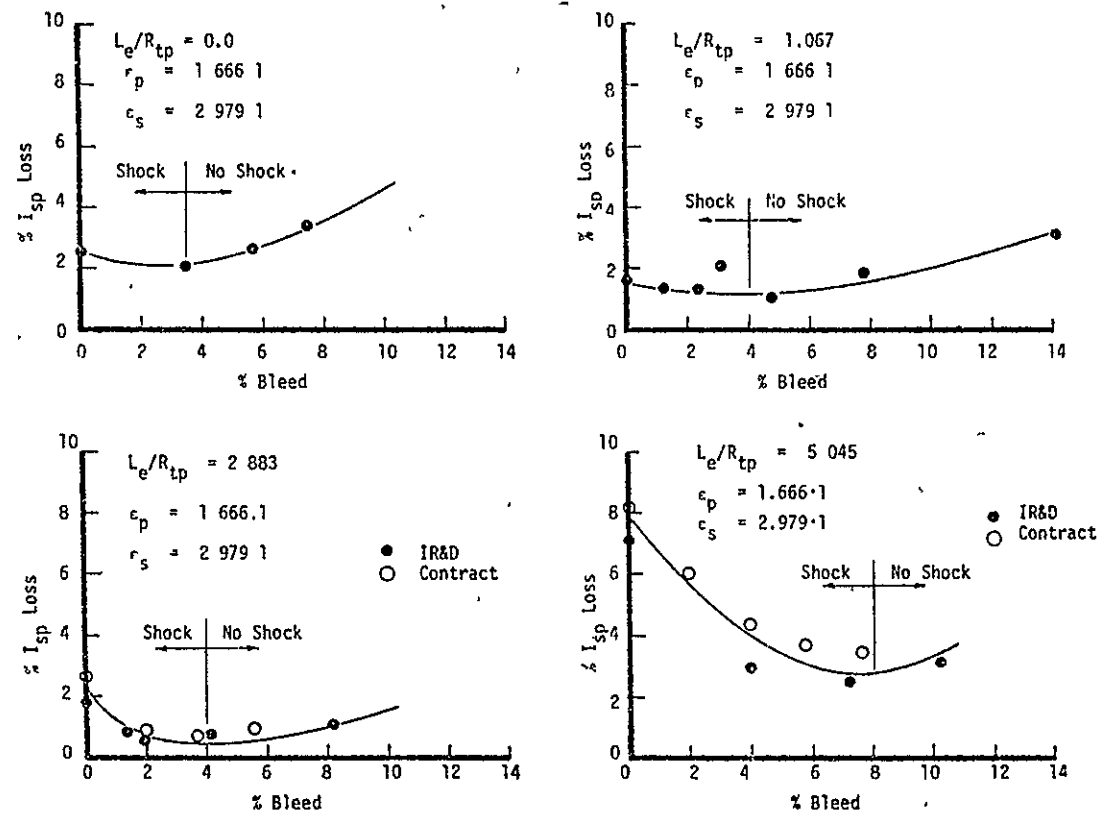


Figure 20. Dual Throat Performance, Relative Nozzle Position Variation

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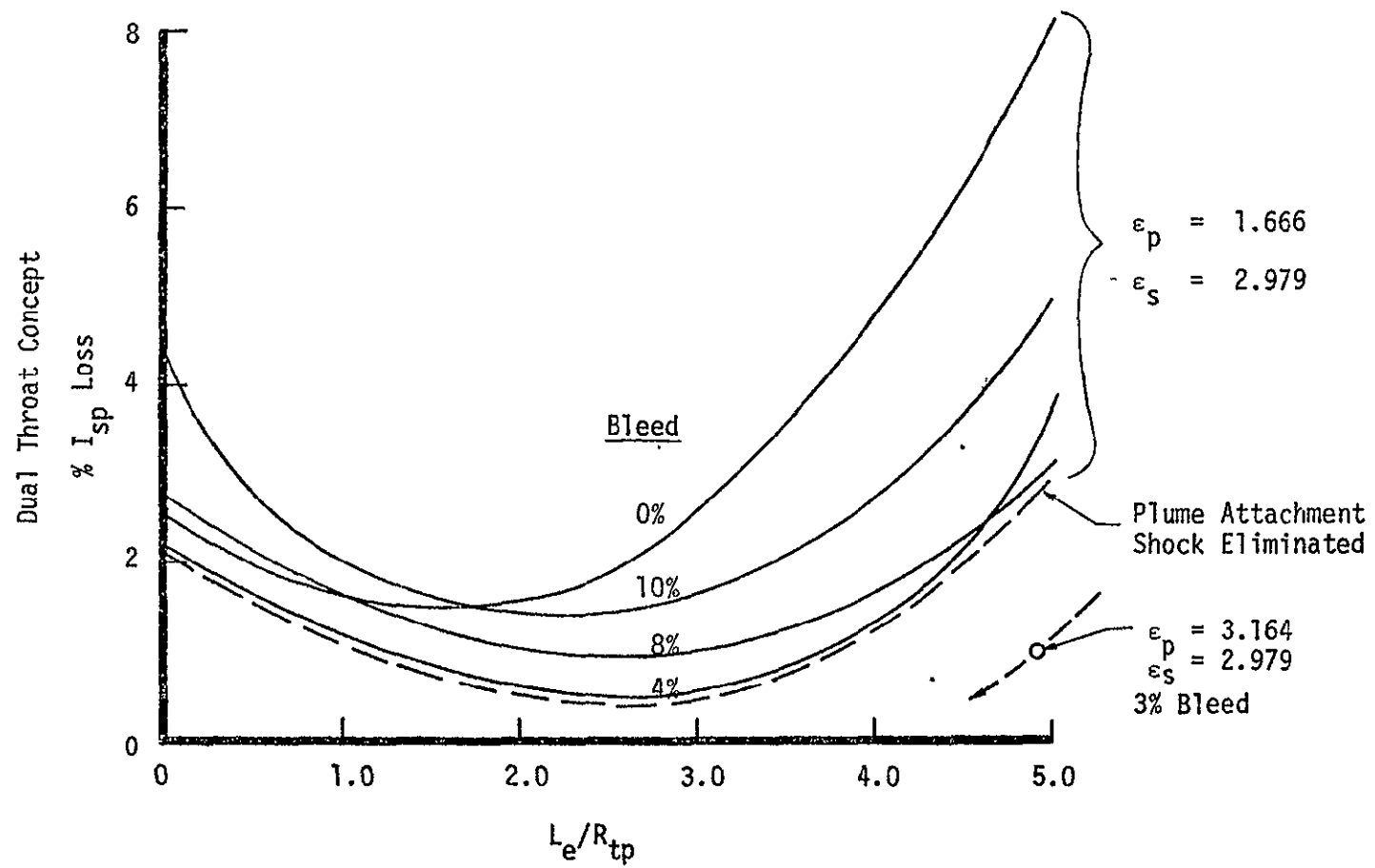


Figure 21. Dual Throat Cold Flow Mode II Performance Loss

TABLE V. - DUAL THROAT COLD FLOW SERIES IA TEST SUMMARY
CONTRACT MODE II TESTS

TEST	TEMPERATURE, °K		CHAMBER STAGNATION PRESSURE, N/cm ²		% BLEED	L _e /R _{tp}	F _{Vac} KN	I _{sp} TEST SEC.	I _{sp} CODE SEC.	% I _{sp} LOSS	ε _p	ε _s
	PRIMARY	SECOND	PRIMARY	SECOND								
101A	363	280	199	8.0	7.7	5.045	3.24	78.5	81.9	3.5	1.666	2.979
102	356	312	202	4.6	0.0	5.045	2.90	74.5	81.9	8.2		
103	369	287	214	5.9	2.0	5.045	3.22	77.6	83.2	6.0		
104	358	285	217	7.0	4.0	5.045	3.39	77.6	81.8	4.4		
105	370	279	223	8.0	5.8	5.045	3.55	79.2	82.8	3.7		
107	366	302	255	3.9	0.0	2.883	3.44	80.2	83.1	2.7		
108	357	286	230	5.7	2.0	2.883	3.63	80.5	81.8	0.8		
109	368	284	234	8.1	5.6	2.883	3.83	81.4	82.7	1.0		
110	356	286	238	7.2	3.7	2.883	3.84	80.4	81.5	0.7		
111	357	278	228	4.8	8.6	4.130	3.70	77.1	81.1	4.3	1.666	1.626
112	349	292	230	2.2	0.0	4.130	3.41	76.0	81.1	5.6		
113	358	282	209	3.9	6.6	4.130	3.35	77.9	81.5	3.7		
114	349	282	210	3.7	5.7	4.130	3.35	77.4	80.7	3.4		
115	367	284	205	3.0	3.5	4.130	3.21	79.3	82.7	3.5		
116	365	285	208	2.7	2.0	4.130	3.20	78.9	82.7	3.9		
117	366	279	210	3.0	6.2	3.934	3.39	79.5	82.3	2.9	3.164	1.626
118	354	288	212	1.1	0.0	3.934	3.25	79.0	81.7	2.6		
119	369	279	208	2.6	4.5	3.934	3.32	80.5	82.9	2.2		
120	363	281	211	2.1	2.3	3.934	3.32	80.6	82.5	1.6		
121	368	280	196	2.7	6.2	3.213	3.16	79.0	82.6	3.6		
122	359	286	199	0.8	0.0	3.213	3.05	79.2	82.2	3.0		
123	374	283	196	2.3	4.5	3.213	3.12	80.2	83.4	3.2		
124	366	285	200	1.7	2.1	3.213	3.13	79.9	82.8	2.8		
125	371	279	195	6.8	8.0	4.930	3.20	79.8	82.7	2.7	3.164	2.979
126	364	285	198	4.7	2.3	4.930	3.14	81.1	82.6	1.1		
127	374	282	197	6.3	6.1	4.930	3.21	81.0	83.2	2.0		
128	366	283	201	5.7	4.2	4.930	3.23	81.0	82.6	1.2		
129	376	277	198	7.6	10.2	4.930	3.29	79.5	82.9	3.5		
130	371	311	201	3.2	0.0	4.930	3.02	79.9	83.5	3.6		

TABLE V. - DUAL THROAT COLD FLOW SERIES IA TEST SUMMARY
CONTRACT MODE II TESTS (cont.)

TEST	TEMPERATURE, °R		CHAMBER STAGNATION PRESSURE, PSIA		BLEED	L_e/R_{tp}	F_{Vac} LBF	I_{SP} TEST SEC	$I_{SP,ODE}$ SEC	% I_{SP} LOSS	ϵ_p	ϵ_s
	PRIMARY	SECOND.	PRIMARY	SECOND.								
101A	654	504	288	11.6	7.7	5.045	728	78.5	81.9	3.5	1.666	2.979
102	640	561	293	6.7	0.0	5.045	653	74.5	81.9	8.2		
103	664	517	310	8.6	2.0	5.045	724	77.6	83.2	6.0		
104	645	513	315	10.2	4.0	5.045	761	77.6	81.8	4.4		
105	665	503	323	11.6	5.8	5.045	799	79.2	82.8	3.7		
107	659	543	327	5.6	0.0	2.883	774	80.2	83.1	2.7		
108	642	514	333	8.3	2.0	2.883	817	80.5	81.8	0.8		
109	663	511	340	11.8	5.6	2.883	862	81.4	82.7	1.0		
110	640	514	345	10.4	3.7	2.883	863	80.4	81.5	0.7		
111	642	501	330	6.9	8.6	4.130	832	77.1	81.1	4.3	1.666	1.626
112	628	525	334	3.2	0.0	4.130	766	76.0	81.1	5.6		
113	644	507	303	5.7	6.6	4.130	754	77.9	81.5	3.7		
114	629	508	304	5.4	5.7	4.130	753	77.4	80.7	3.4		
115	660	511	298	4.4	3.5	4.130	722	79.3	82.7	3.5		
116	657	513	302	3.9	2.0	4.130	720	78.9	82.7	3.9		
117	658	502	304	4.4	5.2	3.934	761	79.5	82.3	2.8	3.164	1.626
118	638	518	308	1.6	0.0	3.934	730	79.0	81.7	2.6		
119	665	503	301	3.8	4.5	3.934	747	80.5	82.9	2.2		
120	654	506	306	3.0	7.3	3.934	747	80.6	82.5	1.6		
121	662	504	284	3.9	6.2	3.213	710	79.0	82.6	3.6		
122	646	514	289	1.2	0.0	3.213	685	79.2	82.2	3.0		
123	674	509	284	3.3	4.5	3.213	701	80.2	83.4	3.2		
124	658	513	290	2.5	2.1	3.213	704	79.9	82.8	2.8		
125	667	503	283	9.9	8.0	4.930	720	79.8	82.7	2.7	3.164	2.979
126	655	513	287	6.8	2.3	4.930	706	81.1	82.6	1.1		
127	673	507	286	9.1	5.1	4.930	722	81.0	83.2	2.0		
128	658	510	291	8.3	4.2	4.930	726	81.0	82.6	1.2		
129	677	499	287	11.0	10.2	4.930	740	79.5	82.9	3.5		
130	667	560	291	4.7	0.0	4.930	680	79.9	83.5	3.6		

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III, D, Data Analysis (cont.)

The remaining 20 tests were conducted using different primary and secondary nozzle area ratio combinations. The primary objective of these tests was to determine the Mode II performance effects produced by different geometry configurations and bleed flows. The wall pressure profiles and corresponding performance losses are plotted in Figures 22 to 25.

The same trends in performance loss and shock elimination are evident with these data as were evident in the Series I results. The summary of these data is shown in Figure 26. These test results show that the performance losses are minimized by increasing the primary nozzle exit area and decreasing the secondary throat area, indicating that the plume attachment shock strength and the resultant performance loss are reduced as the gap between the nozzles is eliminated. In the limit, of course, the dual throat losses go to zero as the gap is closed forming a conventional nozzle.

2. Series II - Secondary Blockage Test Results

The performance loss determined by the Series I testing is restricted to Mode II operation of the dual throat engine. In practice, the compatibility of Mode I and Mode II operation must be considered when designing an engine. Therefore, Mode I secondary blockage (Series II) tests were conducted during the IR&D effort.

For all Series II tests 100% of the flow emanated in the secondary nozzle. The objectives and approach for this series are given in Figures 2 and 27.

Five tests were completed, one with the primary nozzle removed, and four other tests with the primary nozzle at discrete positions. Table VI is a summary of the tests conducted during this series.

Figures 28-30 show the measured wall pressure distribution along the secondary wall for three primary nozzle positions, respectively. Also included in each figure, for comparison, is the wall pressure distribution obtained when the primary nozzle was completely removed.

The tests were run with a constant secondary flow rate of about 6.4 Kg/s (14 lbm/sec). As the primary nozzle is located further from the secondary throat, the actual flow area increases thus reducing the plenum chamber pressure.

The secondary blockage test results indicate that blockage of the primary nozzle becomes a problem if the separation distance is too small and the primary nozzle begins to obstruct the flow in the secondary

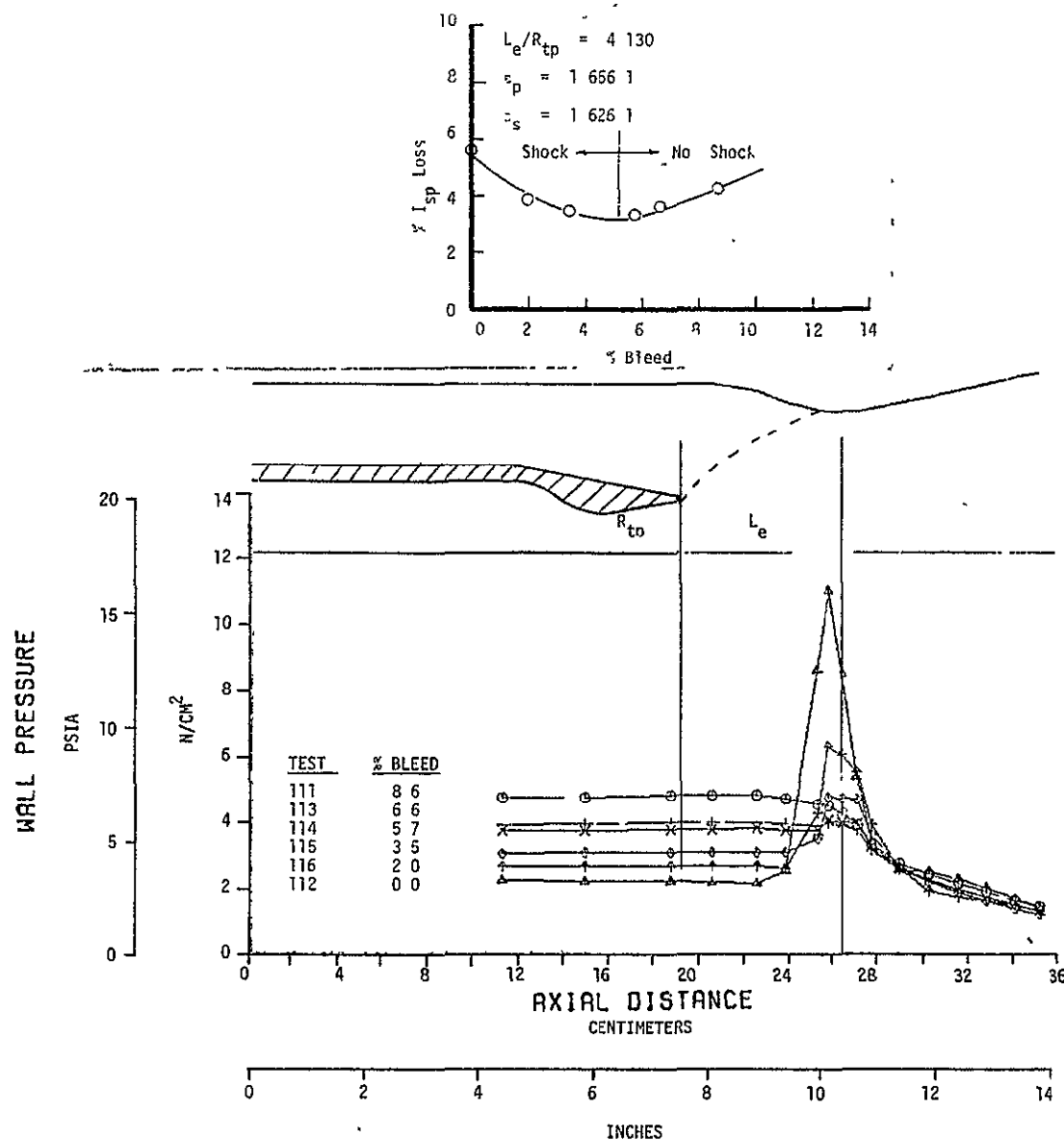


Figure 22. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{TP} = 4.130$)

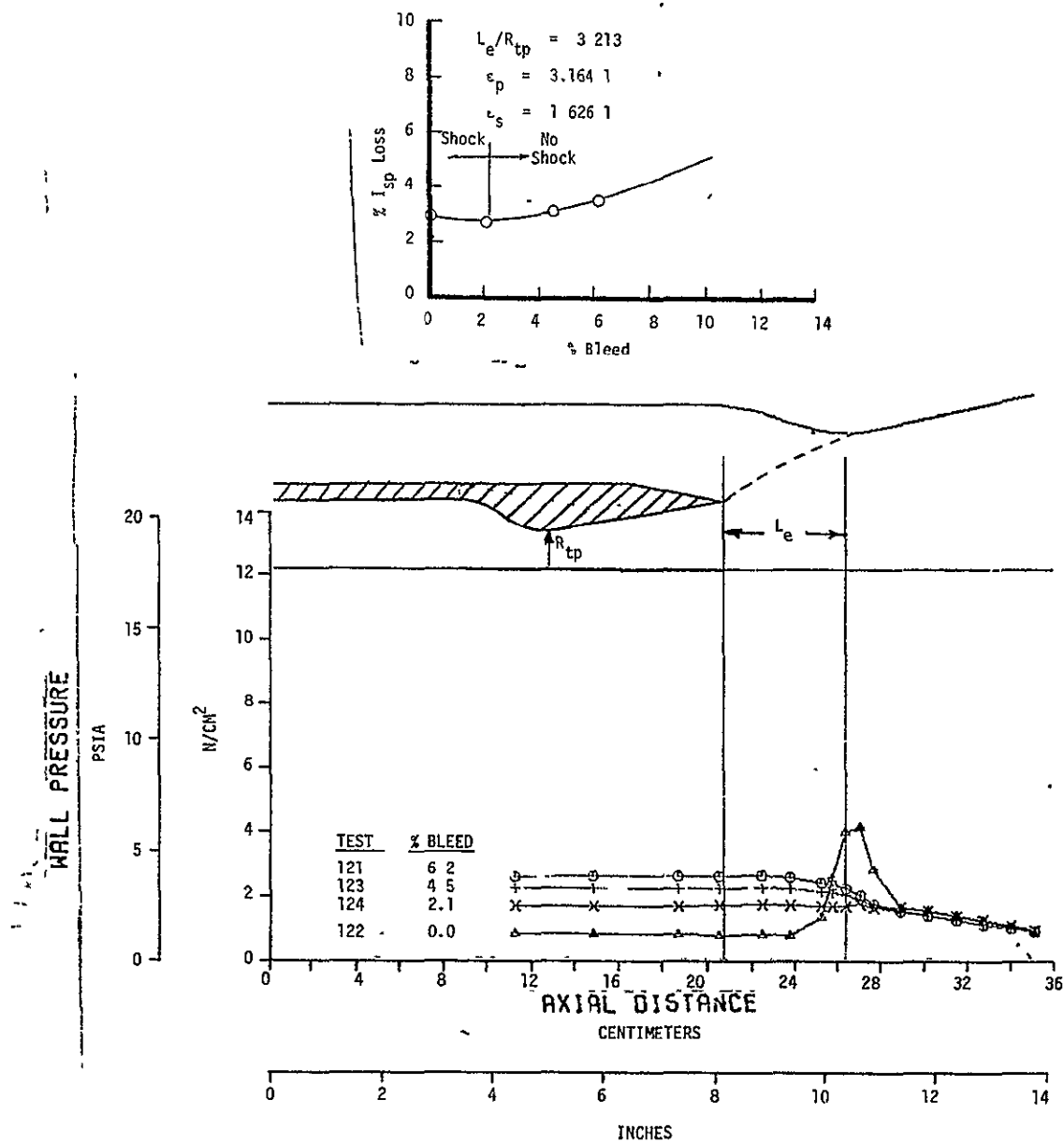


Figure 23. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{tp} = 3.213$)

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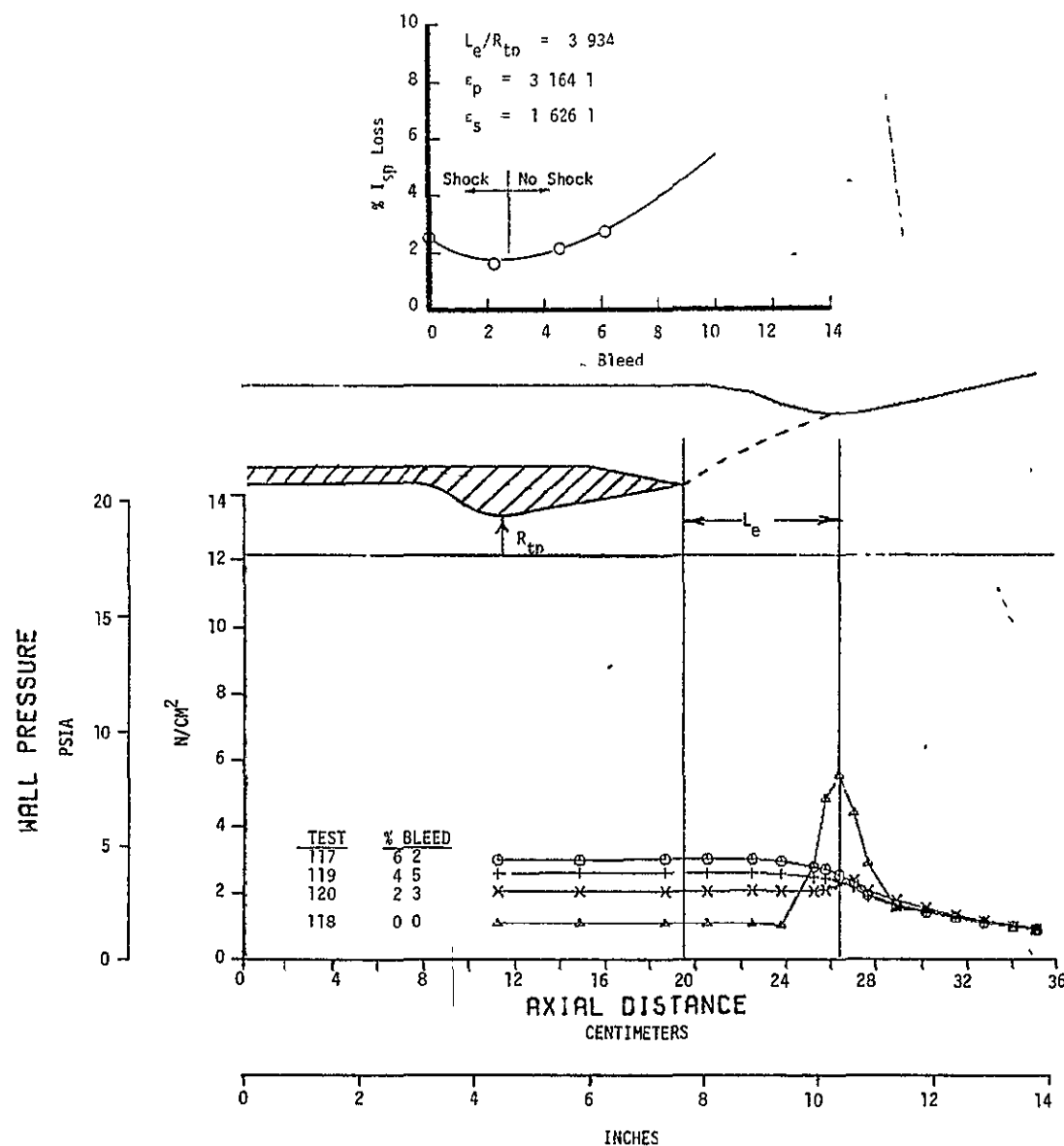


Figure 24. Dual Throat Cold Flow Mode II Plume Attachment
Test ($L_e/R_{TP} = 3.934$)

WALL PRESSURE

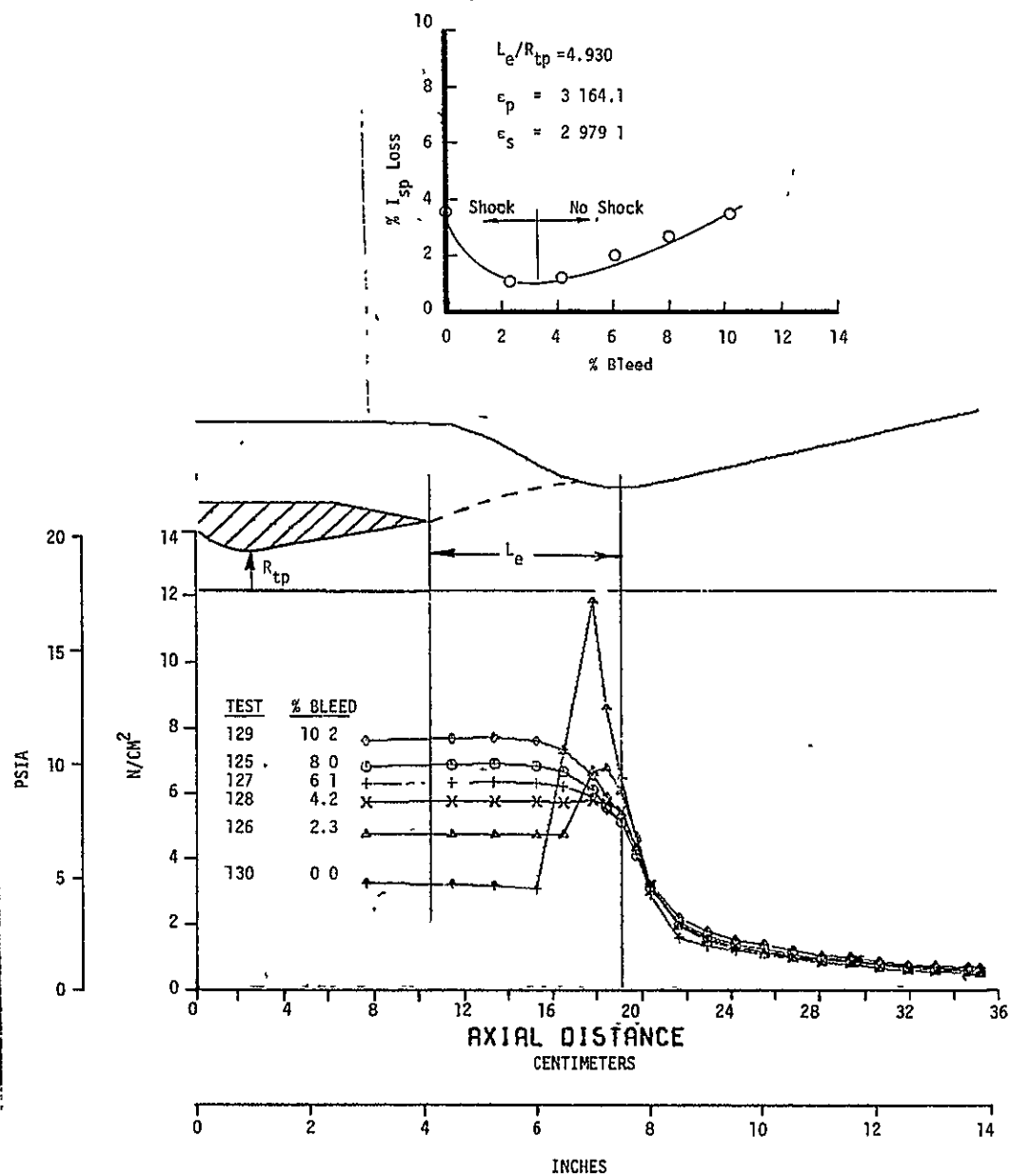
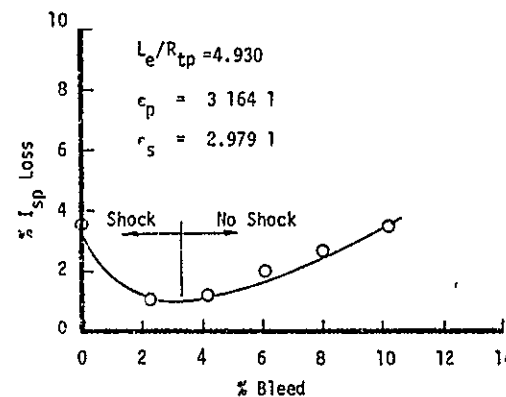
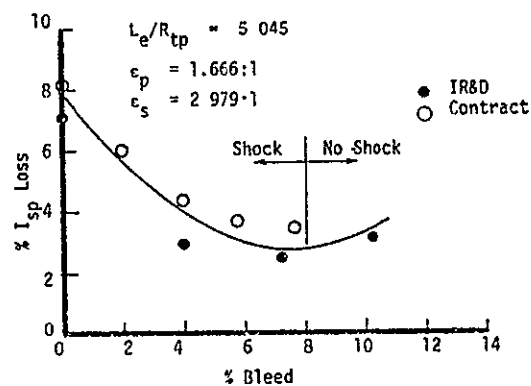
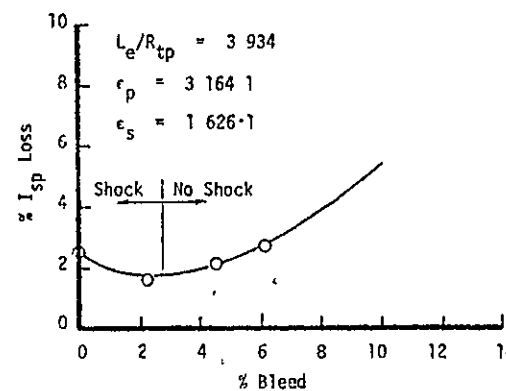
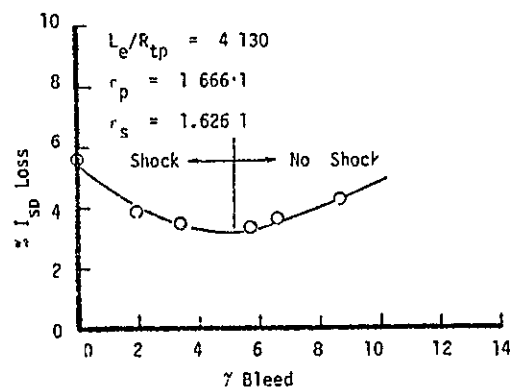


Figure 25. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{TP} = 4.930$)

SMALLER SECONDARY THROAT DIAMETER
LOWER PERFORMANCE LOSS

LARGER PRIMARY AREA RATIO
SMALLER "GAP"
LOWER PERFORMANCE LOSS



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Figure 26. Dual Throat Performance, Nozzle Area Ratio Variation

● OBJECTIVES

- INFLUENCE OF PRIMARY NOZZLE LOCATION ON SECONDARY FLOW
- LOW EXPANSION LOSS

● APPROACH

- REMOVE PRIMARY
- VARIABLE NOZZLE LOCATION
- SECONDARY WALL PRESSURE MEASUREMENT

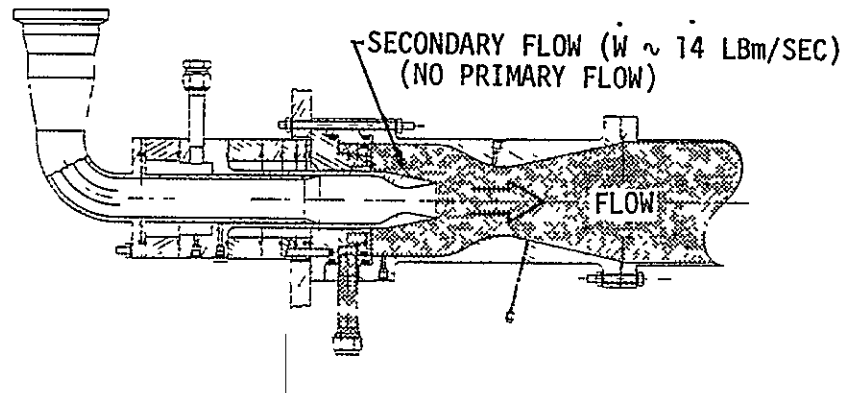


Figure 27. Series II Secondary Blockage Test (No Primary Flow) Program

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TABLE VI
SECONDARY BLOCKAGE SUMMARY

<u>TEST NO.</u>	<u>SECONDARY FLOW</u>		<u>P_{CS}</u>		<u>L_e/R_{tp}</u>
	<u>Kg/s</u>	<u>(lbm/sec)</u>	<u>N/CM²</u>	<u>(psia)</u>	
124	6.21	(13.7)	25.3	(36.7)	Removed
125	6.26	(13.8)	29.1	(42.2)	5.045
126	6.21	(13.7)	28.6	(41.54)*	2.883
127	6.40	(14.1)	43.8	(63.5)	1.067
128	6.26	(13.8)	54.4	(78.9)	0.000

*DATA QUESTIONABLE (DIFFUSER FLOW ATTACHMENT?)

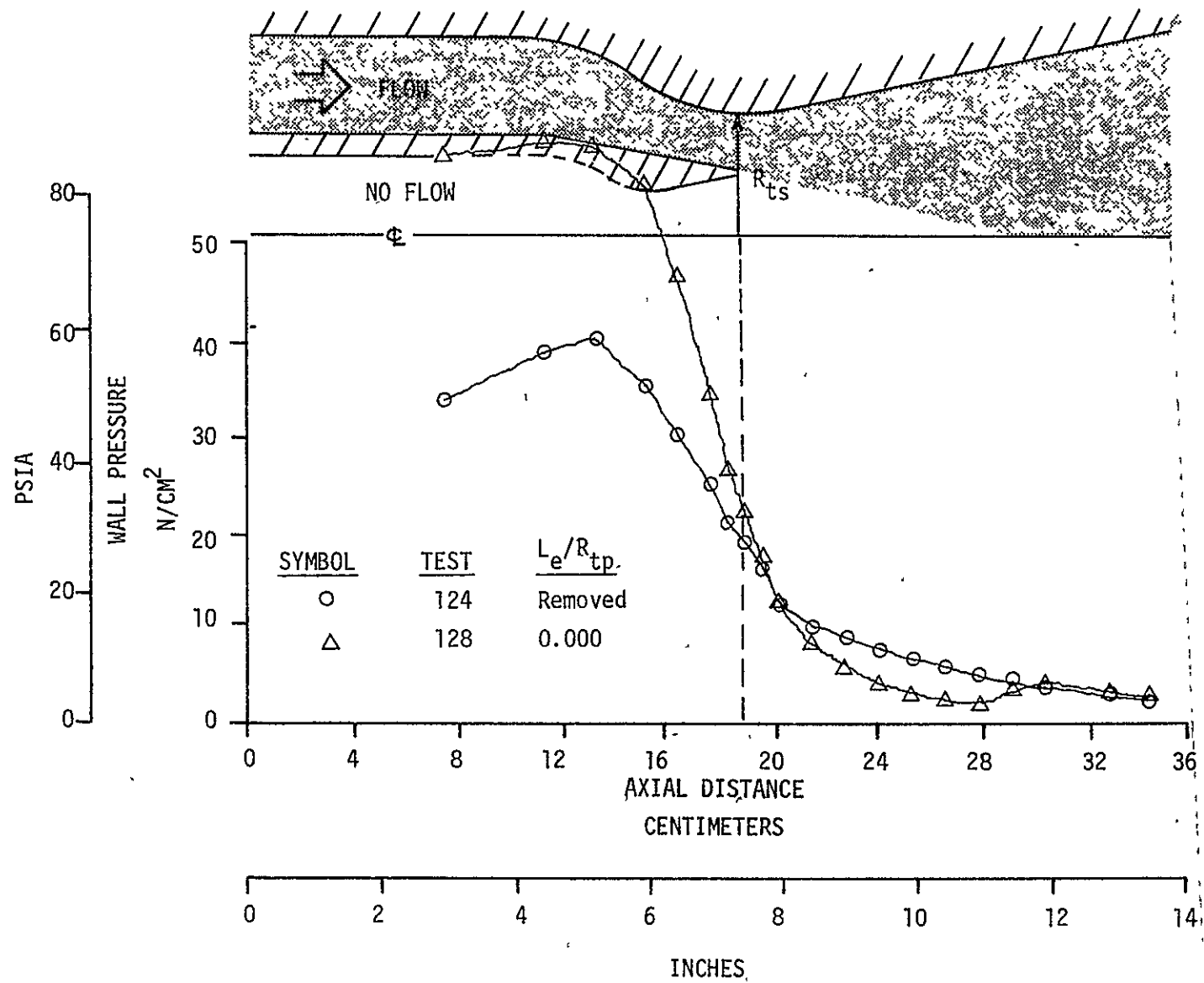


Figure 28. Secondary Blockage Tests $L_e/R_{TP} = 0.000$

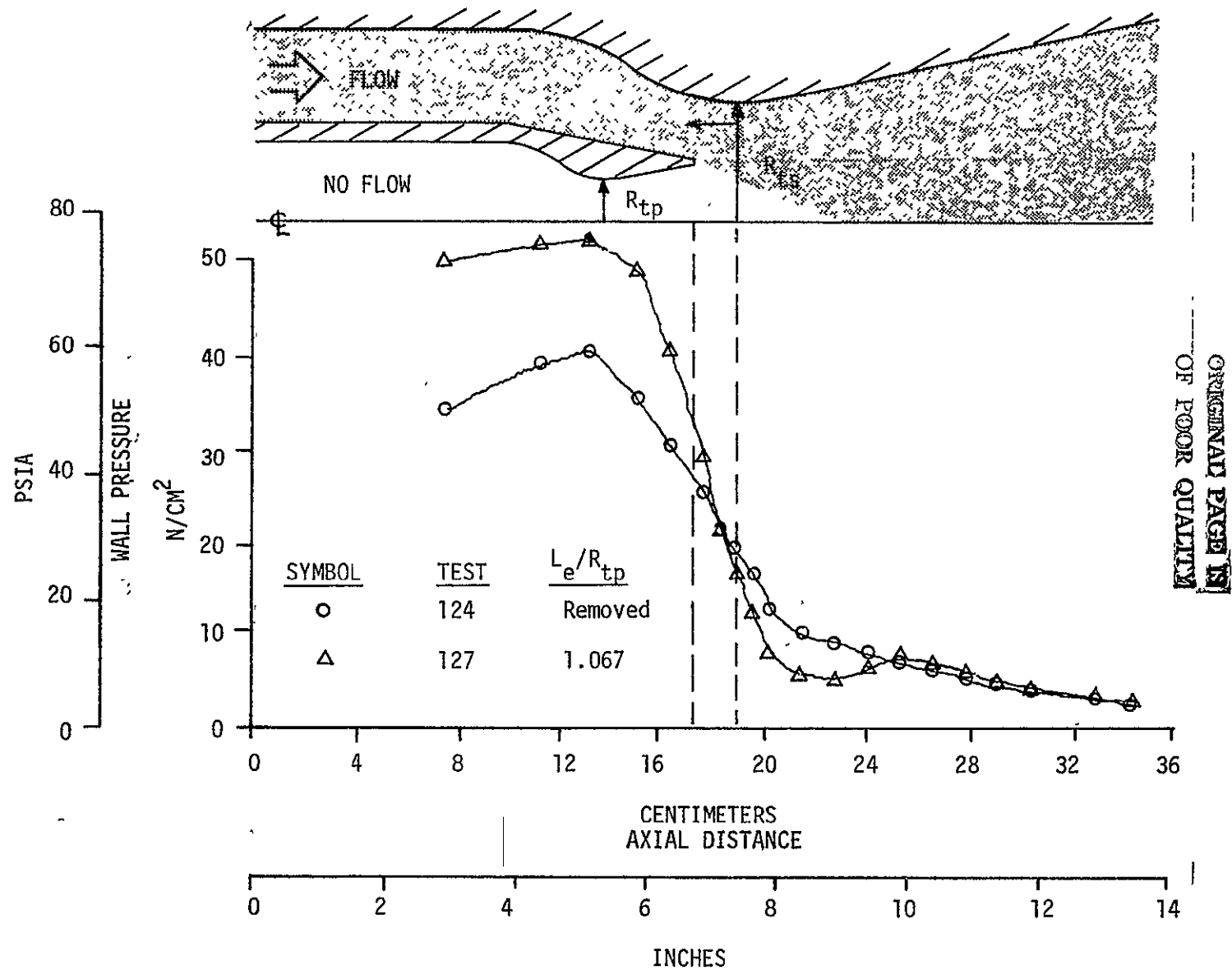


Figure 29. Secondary Blockage Tests $L_e/R_{TP} = 1.067$

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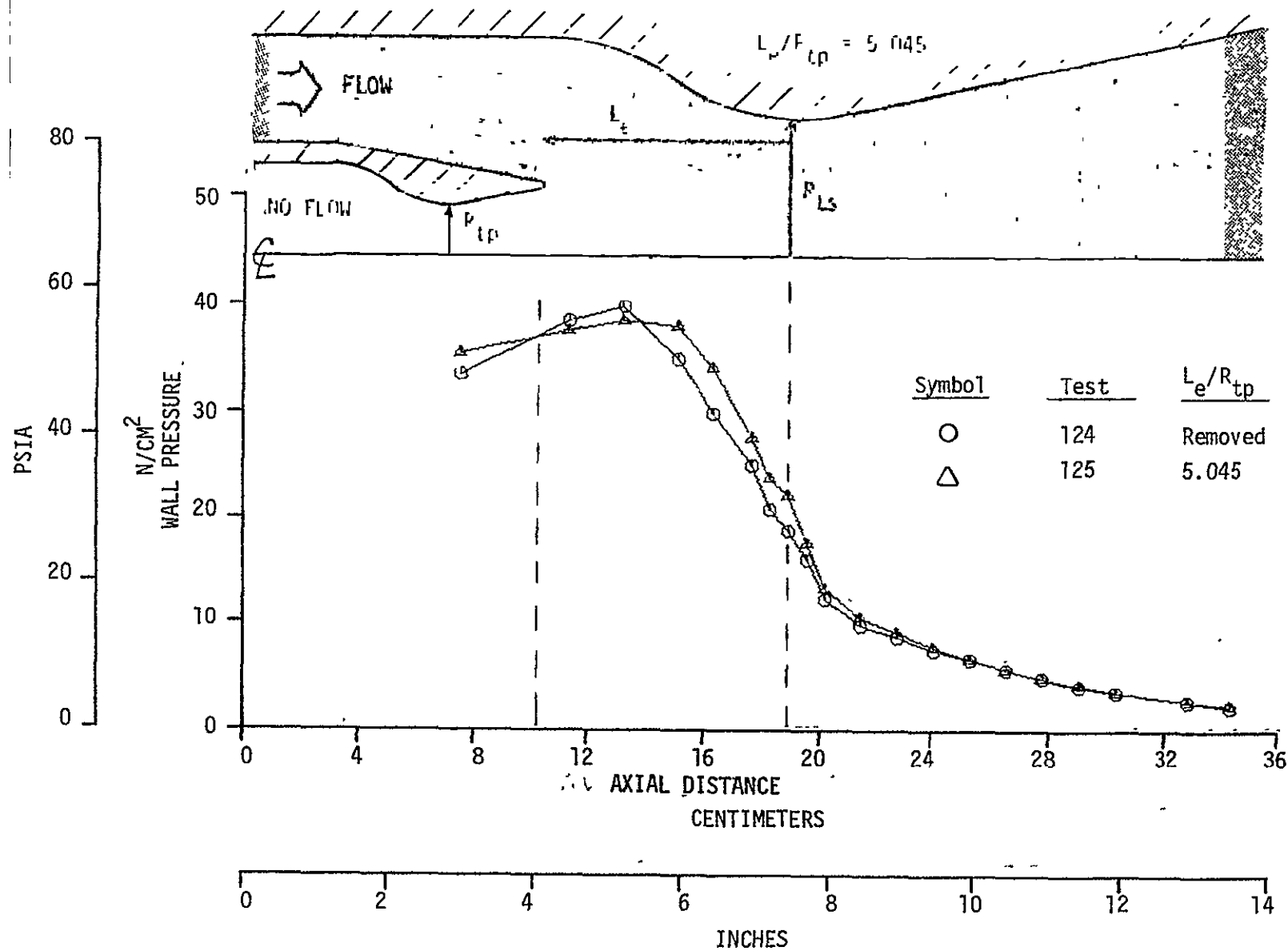


Figure 30. Secondary Blockage Tests $L_e/R_{TP} = 5.045$

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III, D, Data Analysis (cont.)

nozzle. To maintain choked flow in the secondary circuit solely at the secondary throat plane, the primary nozzle must be located such that the flow area between the primary and secondary nozzles is greater than the secondary throat area. Figure 31 is a plot of the flow area (A_F) divided by the secondary throat area (A_{TS}) versus the nozzle separation distance L_e/R_{tp} for both the low ($\epsilon = 1.666:1$) and high ($\epsilon = 3.164:1$) area ratio primary nozzles. To maintain a flow area ratio $A_F/A_{TS} \geq 1.0$, the L_e/R_{tp} must be >1.9 for the low area ratio nozzle and >2.5 for the high area ratio primary nozzle. The Mode I and Mode II aerodynamic operating regions are shown plotted together in Figure 32.

3. Series III - Parallel Flow Test Results

In this series of tests both nozzles were operated with sonic flow through their respective throats. The flow split (% primary/% secondary) ranged from 10%/90% to 30%/70%. The test objectives and approach are given in Figure 33.

A summary of the parallel tests is listed in Table VII. The primary chamber pressure ranged from 34-79 N/cm² (50-115 psia). The secondary chamber pressure from 6.7-41 N/cm² (10-60 psia). The chamber pressure ratio (P_{cs}/P_{cp}) ranged from .311 to .912. The primary flow ranged from 10-30% of the total flow; the secondary flow ranged from 70-90%.

Of the 14 tests conducted during this series, five tests had sonic flow in both the secondary and primary nozzles. The other tests had subsonic flow in the primary with sonic flow in the secondary. Also shown listed in the table is the normalized primary nozzle flow rate. This term is plotted versus the chamber pressure ratio in Figure 34.

Referring to Figure 34, the normalized flow rate increases to a maximum as the pressure ratio decreases. The maximum constant flow rate corresponds to sonic flow. The range of chamber pressure ratios over which primary sonic flow occurs is $P_{cs}/P_{cp} < 0.7$. The corresponding flow split range is $\dot{w}_p > 19\%$ and $\dot{w}_s < 81\%$. It should be emphasized that these ranges hold only for the particular hardware tested. Upon optimization of the primary nozzle expansion ratio and nozzle half angles the results could vary.

The conclusions of this series of tests are:

- 2. Nominal primary flow to a high back pressure is feasible.

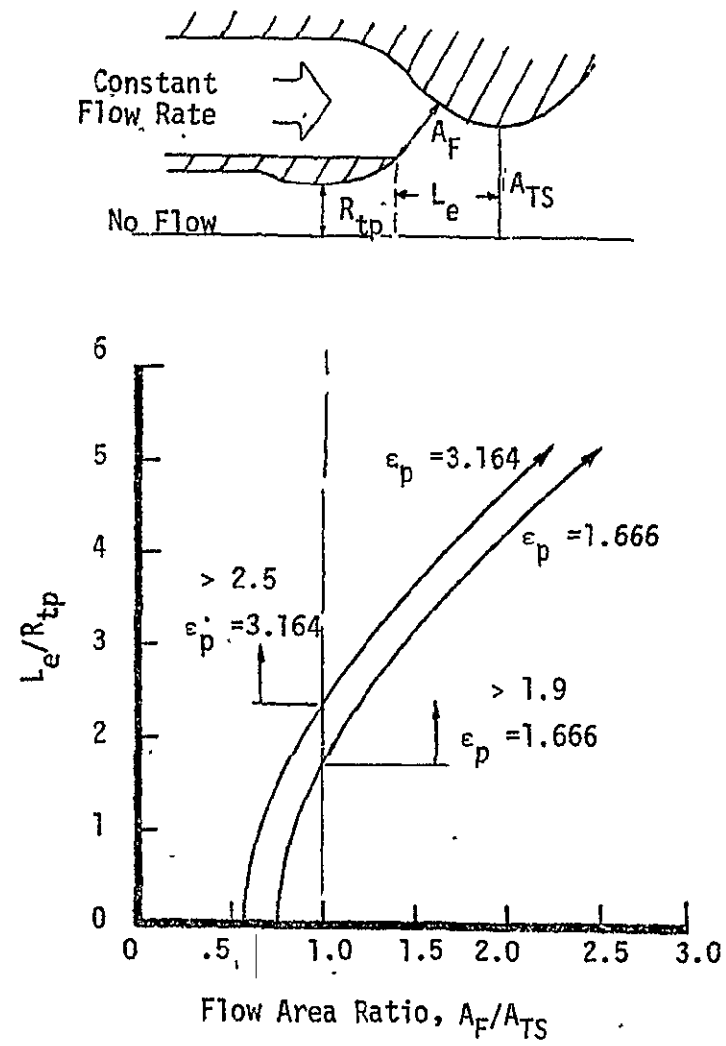


Figure 31. Dual Cold Flow Mode I Blockage Test Results

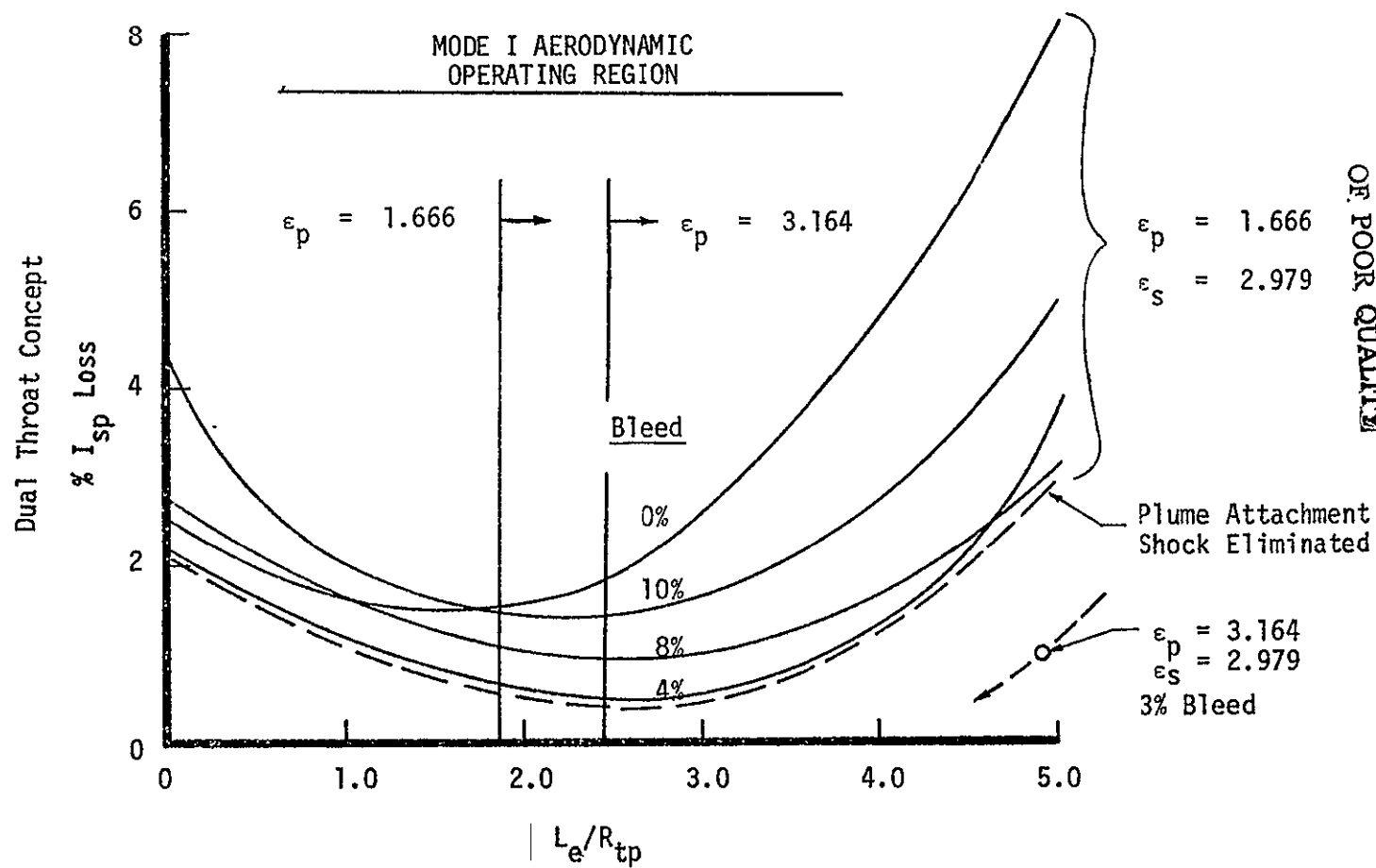


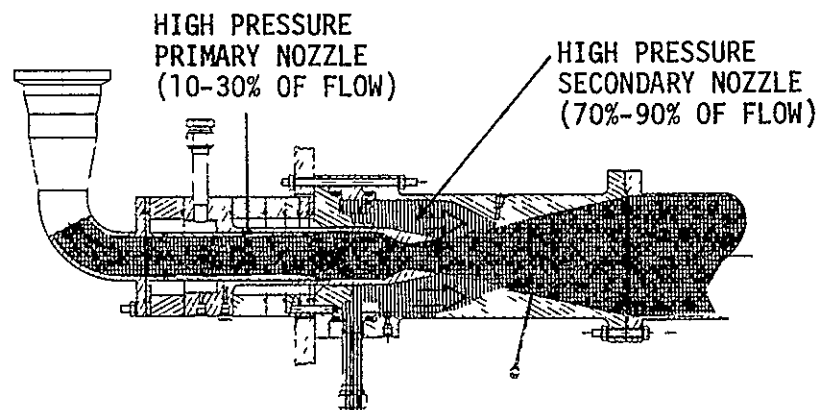
Figure 32. Dual Throat Cold Flow Comparison of Mode I and Mode II Aerodynamic Optimum Operating Regions

● OBJECTIVE

- (1) DEMONSTRATE LIMITS OF ACHIEVING SONIC FLOW IN PRIMARY CIRCUIT WITH VARIOUS BACK PRESSURE
- (2) ESTABLISH RANGE OF CHAMBER PRESSURE RATIOS WHICH WILL RESULT IN SONIC FLOW IN PRIMARY THROAT
- (3) PARALLEL FLOW PERFORMANCE

● APPROACH

- (1) ONE PRIMARY NOZZLE POSITION - $L_e/R_{tp} = 5.045$ (MOST LIMITING CASE - HIGHEST BACK PRESSURE)
- (2) VARY SECONDARY CHAMBER PRESSURE FROM 10 TO 41 N/CM² (15 TO 60 PSIA) AND PRIMARY CHAMBER PRESSURE FROM 38 TO 79 N/CM² (55 TO 115 PSIA)
- (3) MEASURE SECONDARY WALL PRESSURES



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Figure 33. Parallel Flow Test Program

TABLE VII
PARALLEL FLOW SUMMARY

<u>TEST NO.</u>	<u>P_{CP}</u>	<u>P_{CS}</u>	<u>P_{CS}/P_{CP}</u>	<u>% $\dot{W}_p$</u>	<u>% $\dot{W}_s$</u>	<u>$\dot{W}_p \sqrt{T_{CP}/P_{CP}}$</u>	<u>SONIC FLOW</u>
130	56.	12	0.311	100.	0	0.779	*
131	64.	58	0.912	14.	86	0.670	
132	66.	58	0.882	14.	86	0.711	
133	70.	60	0.854	15.	85.	0.747	
134	73.	58	0.800	16	84	0.769	
135	109.	46	0.424	29.	71.	0.778	*
136	65.	58	0.896	13.	87.	0.702	
137	64.	57	0.897	12.	88.	0.679	
138	81.	56	0.693	18.	82.	0.778	*
139	73.	59	0.810	16.	84.	0.771	
140	74.	58	0.788	16.	84.	0.774	
141	68.	59	0.862	17.	83.	0.745	
142	81.	55	0.681	19.	81.	0.780	*
143	113.	46	0.407	29.	71.	0.782	*

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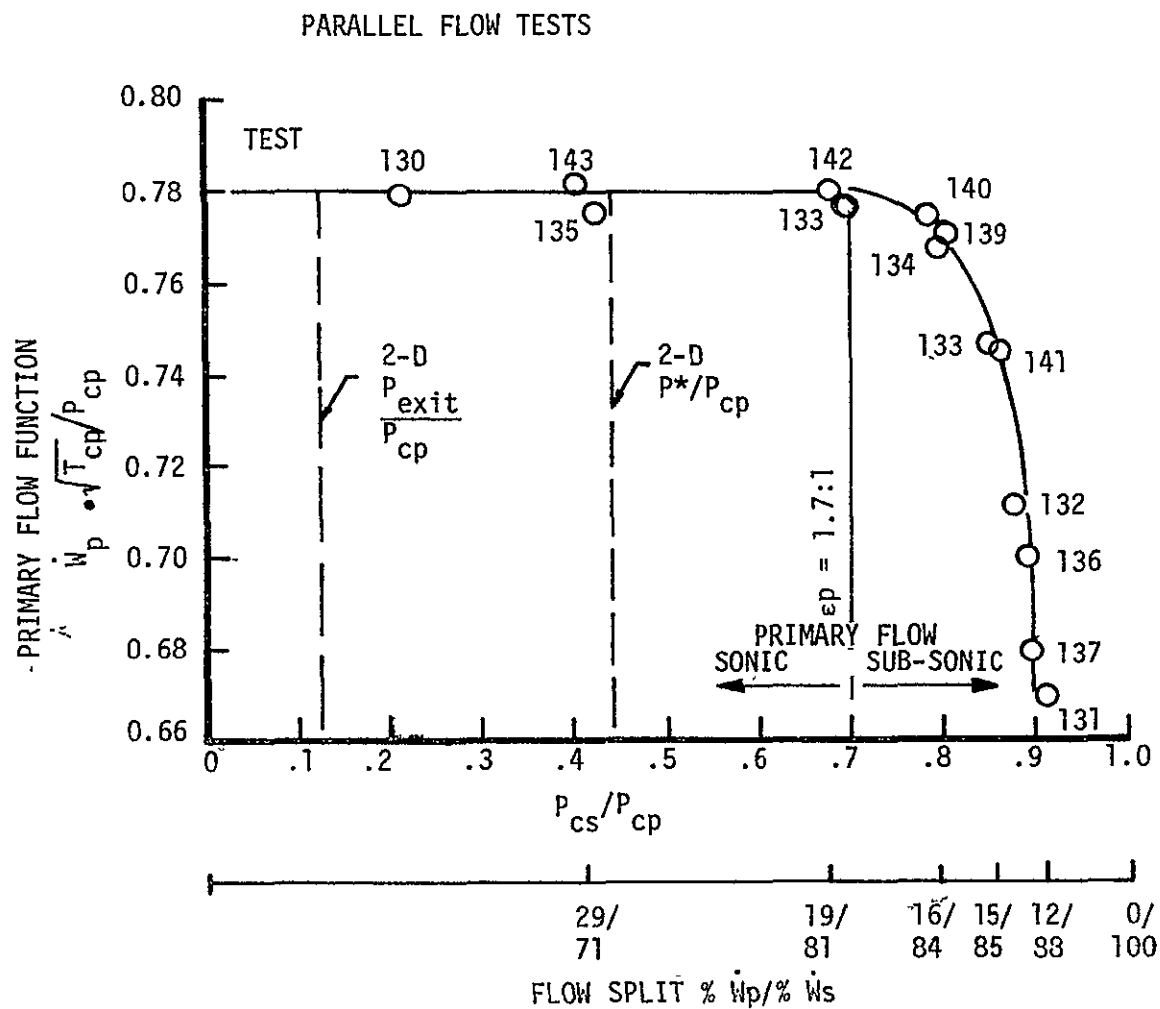


Figure 34. Secondary Flow Function at Various Flow Splits

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III, D, Data Analysis (cont.)

- ° Primary sonic flow was achieved over a range of:
 - ° Chamber pressure ratio $< .7$
 - ° Flow split: $\dot{w}_p > 19\%$ and $\dot{w}_s < 81\%$.

E. CONCLUSIONS

The most important conclusion from these data is that the dual throat concept is aerodynamically feasible. The performance loss obtained with the testing hardware is as low as 0.5 percent. The loss is minimized by an optimum nozzle spacing corresponding to an Af/ATS ratio of about 1.5, or an Le/R_{tp} of 3.0 for the dual throat hardware tested, while requiring only 4% bleed flow. The cold flow results indicate that the Mode I and Mode II geometry requirements are compatible and pose no significant design problems.

IV. AERODYNAMIC MIXING MODEL

A. OBJECTIVES AND GUIDELINES

The primary objective of this task was to develop an analytical model capable of predicting the aerodynamics unique with the Mode II operation of the dual throat thruster.

The plume which is generated by the expanding exhaust gases exiting the primary nozzle and the resultant reattachment to the secondary wall can best be described by two processes: (1) the base pressurization effects caused by the reattachment of the plume to the secondary wall, and (2) the effects of the secondary or bleed flow (mass addition) to the supersonic flow and its effect on the base pressure and plume shape.

A survey of the literature on the above topics revealed the existence of a vast amount of work associated with the base pressurization of the wake associated with high velocity projectiles. Many models have been developed to predict these base pressurization effects, ranging from simple empirical models to highly sophisticated, finite element solutions. Because of the anticipated use of this aerodynamic model for repeated parametric studies in the future, the base pressure theories of Korst, Chapman and Bauer were used in the development of this model. The resulting model met both the resource and technical requirements of this aerodynamic analysis. A complete description of the model and a comparison with the cold flow data are presented in this section.

B. MODELING APPROACH AND DEVELOPMENT

The Mode II (sustainer mode) operation is illustrated in Figure 35. The primary thruster expands the flow to a supersonic condition at the nozzle exit. Further expansion then occurs in the form of a Prandtl-Meyer expansion fan at the nozzle lip. The result is an exhaust plume with a constant pressure boundary. Flow is introduced into the secondary chamber so as to control the location of the plume boundary in order to minimize reattachment shocks which occur when the flow impinges on the secondary nozzle wall. The plume boundary is an exhaust streamline path and acts very much as a nozzle wall. However, a shear layer develops along this boundary due to viscous interaction of the exhaust jet and the gases recirculating in the secondary chamber. Analysis of this shear layer mixing region and the phenomena described above are an important feature of the aerodynamic model.

The assumptions employed in the aerodynamic model are listed below:

- ° The flow leaving the primary nozzle is one dimensional and supersonic.

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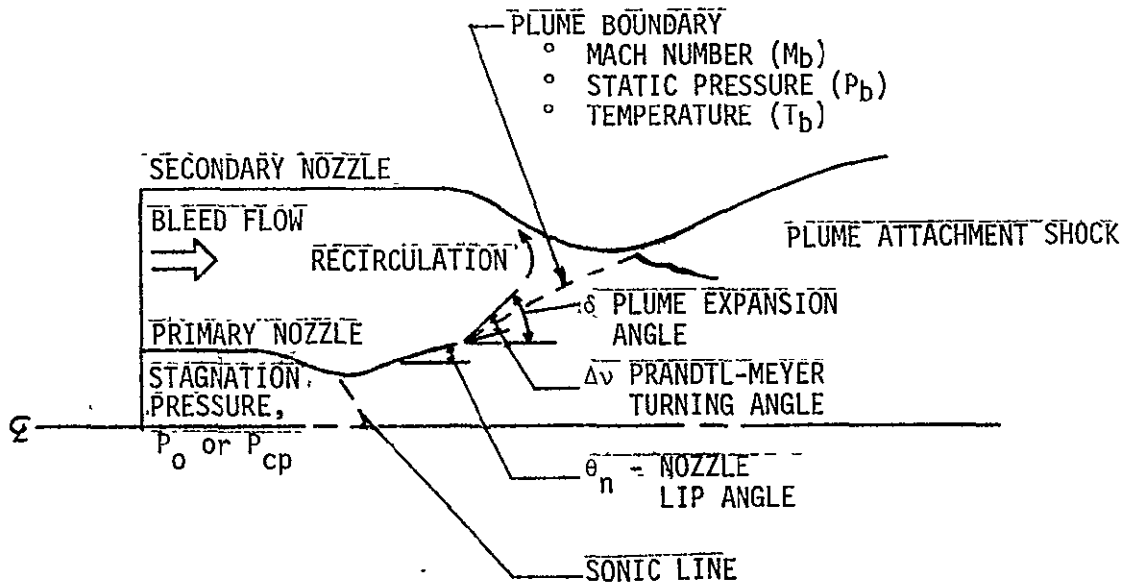
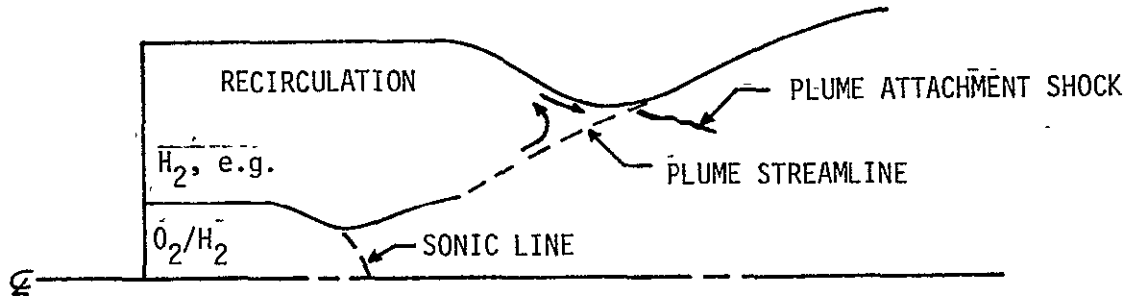


Figure 35. Mode II, Sustainer Mode

IV, B, Modeling Approach and Development (cont.)

- The exhaust is represented by an ideal gas with a constant expansion coefficient, γ .
- The shear layer is represented by the shear layer model developed by Chapman (Ref. 1), Korst (Ref. 2), Bauer (Ref. 3), and others. Assumptions implicit in this model have been retained.
- The shear layer is isoenergetic.
- The shear layer is treated as two dimensional.*
- The shear layer is a constant pressure surface, both in cross section and along its entire length. At the inner edge of the shear layer, the gas velocity is negligible. At the outer edge of the shear layer, the velocity is constant and equal to the velocity of the plume boundary streamline.
- The plume boundary can be represented by the methods developed by Herron (Ref. 4).

Details of the aerodynamic model are discussed below.

1. Nozzle Wall and Plume Geometry

Both the primary and the secondary nozzles are assumed to have geometries that can be described analytically by the following parameters:

- Chamber contraction ratio.
- Circular wall radius connecting the chamber and the nozzle inlet.
- Conical nozzle inlet.
- Circular wall radius upstream of the nozzle throat.
- Circular wall radius downstream of the nozzle throat.

*This assumption has been shown by Bauer (Ref. 3) to be accurate provided that the projected thickness of the mixing zone on the radius of symmetry is less than .3 of the radius of symmetry. This condition is always satisfied in the dual throat engine.

IV, B, Modeling Approach and Development (cont.)

- ° Either a conical exhaust nozzle or a contour shape that can be represented by a spline fit.

An exact definition of these parameters is given in the WALL subroutine. The input data write-up for the program provides the details concerning their use. The above parameters are sufficient to accurately define the geometry of a rocket nozzle. The parameters have been chosen so as to be identical to those used by the TDK program (Ref. 5), which is to be used elsewhere in the performance prediction procedure. The coordinate system of the nozzle geometry is centered at the nozzle throat plane, and the nozzle throat height is assumed to be unity. In this way, a nozzle can be scaled by use of only one number, i.e., the actual nozzle throat height (or throat radius).

The calculations carried out by the aerodynamic model computer program, however, are in the "plume coordinate system". This system is centered at the exit plane of the primary nozzle, and the nozzle exit plane height is assumed to be unity. Thus, the primary and secondary nozzle geometries must be converted to the plume coordinate system. This conversion is done automatically by the computer program.

Once the aerodynamic model calculations have been completed, it is necessary to output an effective nozzle contour for the sustainer nozzle during Mode II operation. This is done for a nozzle operating with sufficient bleed flow that the plume (i.e., shear layer) boundary attaches just downstream of the secondary nozzle throat. Attachment further downstream (or perhaps no attachment) is regarded as requiring an excessive amount of bleed flow. Attachment upstream of the secondary nozzle throat is regarded as unacceptable because of induced shock structure and aerodynamic losses that lower engine performance. The program thus determines a bleed flow rate which is a "best solution" for the operating conditions of the engine.

2. Plume Boundary Shape

A correct shape for a plume boundary caused by a supersonic jet expanding into a quiescent media can be obtained using the method of characteristics (MOC). Computer solutions of this type have been verified to predict plume structure for a wide range of experimental conditions. These MOC solutions require computer calculations that are of the order of several minutes per solution. Thus, they are regarded as not suitable for parametric analysis. Consequently, a number of simpler, correlative techniques have been developed for rapidly predicting plume shape. The basic approach common to these methods is to predict plume shape by matching MOC solutions that have either been published or calculated for the purpose

IV, B, Modeling Approach and Development (cont.)

of providing data to match. A method of this type was selected for incorporation into the aerodynamic model. The method selected to determine the plume boundary shape was that of Herron (Ref. 4). Before a satisfactory choice of method could be arrived at, a number of methods were investigated, programmed, and tried out in the aerodynamic model. For various reasons they were found to be unsatisfactory. Because of the effort expended in determining a proper method, the methods that were investigated are discussed below.

The methods investigated were those of Charwat (Ref. 6), Latvala (Ref. 7), Andes (Ref. 8), and Peterson (Ref. 9). The Charwat method was used in the original Brown Engineering Computer Program, and was therefore immediately available. It was found, however, that the method failed when applied to the operating conditions of the ALRC cold flow test series (Ref. 10). This problem was confirmed by an author of the JANNAF Plume Handbook (Ref. 11). The Charwat method was investigated for inclusion in the handbook, but was omitted when its limitations were found. Instead, the method of Latvala was recommended.

Despite this recommendation, the method of Latvala was found to be highly inaccurate for a gas with an expansion coefficient differing from that for air ($\gamma = 1.4$). For example, the plume shape for an expansion at $\gamma = 1.15$ presented by the Latvala method gives a plume shape much too expansive. The simple correction for values of γ other than 1.4 that is given in the JANNAF Plume Handbook was found to be ineffective. This problem had been noted previously by Andes (Ref. 8) who developed a correlation formula to correct the Latvala plume shape for a broad range of conditions. The Andes method, however, was found to be not applicable to the Latvala method as given in the JANNAF Plume Handbook and required use of a computer program developed by Andes. This program was found to be no longer available.

The method of Peterson was also investigated. This method, which appears to be quite accurate, is contained in an appendix to Reference 9. It has the shortcoming of being available only for flows with $\gamma = 1.15$. Working out the charts on which the method is based for other expansion coefficients would be time consuming and would require a set of MOC calculations to be carried out. Because the method is essentially geometrical and also requires a set of charts, it is not readily suitable for use as a computer subroutine. Unlike the Latvala method, the Peterson methods use a physically correct choice of correlation parameters.

It was thus necessary to examine other methods for predicting plume shape. Next, the Herron method was tried and found to be satisfactory when suitably modified. This method uses the quantity M_b/γ as a similarity parameter to simplify the problem. As defined in the

IV, B, Modeling Approach and Development (cont.)

Nomenclature, M_b is the Mach number along the plume boundary and γ is the ratio of heat capacities. Given the plume expansion angle at the nozzle lip, defined as the nozzle lip angle plus the Prandtl-Meyer expansion angle, i.e., $\delta = \theta_n + \Delta v$, and given the parameter M_b/γ , three charts can be used to locate three fixed points along the plume boundary. These points correspond to $x/r_e = 5, 10$ and 18 . For example, given $x/r_e = 5$, and values of δ and M_b/γ , the charts can be used to determine a value for r/r_e . Herron has correlated this method against numerous plume shapes, ranging from δ 's of 45° through 65° and M_b/γ values of $5, 7, 10$, and 13 .

Unfortunately, the available charts are so small as to be almost unreadable. Also, for the dual throat nozzle, the plume shapes of interest are for values of x/r_e that are generally less than five, values of M_b/γ less than five, and values of δ less than 45° . Like the other methods investigated, the Herron method is intended to approximate plume shapes where the plume is highly under expanded and, thus, relatively large in extent.

It was necessary, therefore, to extend the Herron method to the region of interest for the dual throat nozzle. This was done by carrying out a series of MOC plume calculations and developing a set of tables from the results. The plume calculations were carried out so that tables could be constructed over the range:

$$\begin{aligned}M_b/\gamma &= 2, 3, 5, 7, 10, \text{ and } 13 \\ \delta &= 30, 45, 60, 75^\circ \\ x/r_e &= 3, 5\end{aligned}$$

Next, it was necessary to determine a plume shape from the points obtained from the correlation method. It has been found that a plume boundary can be approximated by a circular arc with reasonable accuracy (Refs. 7 and 8). Three conditions are required to determine a circle, and thus, several choices are possible. For example, the Latvala method, as presented in the Plume Handbook (Ref. 11) uses the nozzle exit coordinates, the angle δ , and the radius of the circle as the three conditions. A better choice in matching plume shape, however, was found to be three points as follows:

$$(1,0), (r/r_e|_3,3), \text{ and } (r/r_e|_5,5).$$

In this way the plume is forced to pass through three points in the region of interest, although in general the plume expansion angle at the nozzle lip will not be satisfied.

3. Shear Layer Model and Its Use in the Aerodynamic Model

The shear layer model is essentially as described by Bauer in Reference 3. The shear layer is assumed to begin at the exit

IV, B, Modeling Approach and Development (cont.)

of the primary nozzle where its thickness is negligible. It then develops as a turbulent shear layer along the boundary of the primary nozzle exhaust plume. Viscous interaction of the exhaust plume with gases recirculating in the chamber of the secondary nozzle form the shear layer. In steady state operation, the system will maintain a stable equilibrium base pressure. This base pressure determines the plume boundary. The theory assumes that a streamline divides the shear layer such that gases on one side of the streamline are recirculated into the base region. This is the so called "dividing streamline" concept. At steady state operation, all of the gases exhausting from the primary nozzle will also exhaust from the secondary nozzle. Gases in the recirculating region (in this case the chamber of the secondary nozzle) have been trapped because they are degraded in total pressure to the extent that they cannot penetrate the static pressure rise produced by turning of the flow at the point where the shear layer impinges the secondary nozzle wall. It is assumed in this model, that the dividing streamline becomes a stagnation point at the secondary wall. This stagnation occurs abruptly either by a compression that is essentially isentropic, or by a non-isentropic compression through a shock structure. Details of this process govern the value that will be obtained for a base pressure, i.e., the pressure along the plume boundary. Methods by which this base pressure value can be estimated are discussed in Section b, below.

Assuming that the base pressure is known from experimental data for the case of no bleed flow, it is possible to completely determine the flow conditions in the nozzle, both with and without bleed flow, using the procedures outlined in the following steps:

(1) Conditions at the exit of the primary nozzle are determined using the one-dimensional relation between Mach number and area ratio, i.e.,

$$\epsilon_p = M_e^{-1} \left[\frac{2}{\gamma+1} \left(1 + \frac{\gamma-1}{2} M_e^2 \right) \right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

(2) The exhaust flow is turned at the nozzle lip and expands to equal the boundary Mach number M_b . The angle of this turn, Δv , is determined using the Prandtl-Meyer relations for supersonic expansion at a point, as follows:

$$\Delta v = v_b - v_e$$

IV, B, Modeling Approach and Development (cont.)

where:

$$v_b = \left(\frac{\gamma+1}{\gamma-1}\right)^{1/2} \arctan \left[\frac{\gamma-1}{\gamma+1} (M_b - 1)\right]^{1/2} - \arctan [M_b^2 - 1]^{1/2}$$

$$v_e = \left(\frac{\gamma+1}{\gamma-1}\right)^{1/2} \arctan \left[\frac{\gamma-1}{\gamma+1} (M_e - 1)\right]^{1/2} - \arctan [M_e^2 - 1]^{1/2}$$

(3) The plume boundary is determined using the modified method of Herron as previously discussed. Since the plume boundary is a circular arc, all of its geometric properties (such as its position, arc length to any point, and slope at any point) are known analytically.

(4) The intersection of the plume with the secondary nozzle wall is determined geometrically.

(5) The Korst method allows the shear layer properties to be determined exactly at the point of intersection of the plume boundary and the wall. All of these properties can be evaluated from various integrals of the error function, as described in the next section.

(6) The plume boundary is a stream line. This stream-line will be turned abruptly (i.e., deflected) to follow the secondary nozzle wall, resulting in a shock structure. The cases of most interest correspond to solutions of the shock relations of the weak family. Cases where a lambda shock structure develops, so that the flow is shocked subsonic in part of the region, are of less interest since they correspond to aerodynamically poor designs. The oblique shock relations are used to determine the rise in static pressure across the shock and the loss of total pressure across the shock. Since the development of shock structure in the nozzle is fundamentally a two-dimensional phenomenon, no attempt is made to treat the flow in the nozzle downstream of the origin of the shock.

(7) Once a solution has been obtained for the zero bleed flow case, the calculations are repeated using successively larger values for the base pressure. The bleed flow rate that corresponds to this pressure increase is then evaluated using the method presented by Bauer (Ref. 3) and given in the following section. The calculations terminate when the plume boundary is found to attach downstream of the secondary nozzle throat.*

*A more exact description of steps (1) through (7), above, is presented in Appendix A, the problem statement for the aerodynamic model.

IV, B, Modeling Approach and Development (cont.)

The steps described above are sufficient to determine the shape of the "wall" that bounds the exhaust from the primary nozzle during Mode II operation. In order to determine the aerodynamic performance of one of these solutions, it is necessary to input this wall to the TDK computer program. The TDK program gives a MOC solution for the nozzle. Shocks are ignored by TDK, but an optimum design will have weak shocks that can be ignored, i.e., treated as isentropic compression. The interface between the shear layer model and TDK has been automated by developing a spline curve fit to the primary exhaust "wall" shape, i.e., the primary nozzle, plume boundary, and the secondary nozzle contour downstream from the plume attachment point.

a. Governing Equations for the Shear Layer Model

A derivation of the shear layer model has been given in numerous reports such as References 2, 3, and 12. Much material and an exhaustive list of references on the subject are presented in Reference 13. Therefore, only a summary of the equations used will be given here.

At a distance, x_j , along the shear layer as measured from its origin, a complete solution is available for properties transverse to the shear layer in terms of a non-dimensional distance, η .

A dimensional distance, Y , can be obtained from η as follows:

$$Y = \eta x_j / \sigma$$

where:

σ is the "shape factor" of the shear layer.

Both η and Y are measured from the centerline of the shear layer. The only value of η that is actually required by the model is η_D , the non-dimensional location of the dividing streamline and this value is tabulated as a function of Crocco number in Reference 3.

The velocity profile, ϕ , is given by the relation

$$\phi = [1 + \operatorname{erf} \eta]/2.$$

The Crocco number, Ca_∞^2 , is also required by the method and is simply

$$Ca_\infty^2 = 1 - T_b/T_0$$

IV, B, Modeling Approach and Development (cont.)

where:

$$T_b/T_0 = [1 + \frac{\gamma-1}{2} M_b^2]^{1/2}$$

Two integrals must be evaluated using the above quantities, and these are defined as:

$$(I_0)_\eta \equiv \int_{-\infty}^{\eta} \frac{d\eta}{1 - Ca_\infty^2 \phi^2}$$

$$(I_1)_\eta \equiv \int_{-\infty}^{\eta} \frac{\phi d\eta}{1 - Ca_\infty^2 \phi^2}$$

The subscript is used to designate the upper limit of the integral. These integrals are evaluated using Simpson's rule as the method of integration and a standard subroutine for evaluating the error function. Because of the exponential character of the error function, it is desirable to divide the integral into several intervals. The limit values $\pm\infty$ can be accurately taken as ± 3 , which is the convention used by Bauer (Ref. 3).

Bauer gives an expression for the mass flow between two streamlines that can be applied to relate the pressure and length of the shear layer to a bleed flow rate.

For the case of no bleed flow, the ratio of mass flow entrained on the base flow side of the dividing streamline to the total mass flow is bound to be*:

$$(G/G_T) = M_b \left(\frac{2}{\sigma}\right) \left(\frac{1}{r_p^*}\right)^2 x_j r_w \left(\frac{p_b}{p_o}\right) \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \sqrt{1 - Ca_\infty^2} (I_1)_{\eta_D}$$

If this expression is evaluated for solutions found with a larger base pressure, the difference in mass flow must be due to the bleed flow rate. Therefore, the percent bleed flow associated with these solutions is:

*This expression differs from that given by Bauer (Ref. 3, Equation 13) in that the flow is assumed to be two-dimensional rather than axially symmetric, resulting in a somewhat simplified result.

IV, B, Modeling Approach and Development (cont.)

$$\dot{W}_s^{(i)} / \dot{W}_p = [(G/G_T)^{(i)} - (G/G_T)^{(1)}] * 100$$

where:

$i = 1$ represents zero bleed flow

$i > 1$ represents solutions obtained by raising the base pressure

$\dot{W}_s$ is the weight flow rate in the secondary chamber

$\dot{W}_p$ is the weight flow rate in the primary chamber

It is also necessary to use an expression taken from the literature for the shape factor, σ . Bauer gives two such expressions. The first, due to Korst, is an empirical expression based on experimental data in the Mach number range from 1 to 2, which is

$$\sigma = 12 + 2.758 M_b \quad (\text{Korst})$$

A more basic method presented by Abramovich is preferred by Bauer. The expression is

$$\sigma = 24 (I_1)_3 / (I_0)_3 \quad (\text{Abramovich})$$

Both expressions give the classical result that for incompressible flow ($M_b = 0$) the shape factor is:

$$\sigma_{ic} = 12$$

It was found that the aerodynamic model successfully correlated the ALRC cold flow data when the Abramovich model was used, but bleed flow vs base pressure was over predicted somewhat when the Korst model was used. This trend is because the Abramovich formulation gives lower values for σ than the Korst method (see Figure 3 of Ref. 3).*

Reference 13 gives data and models for about a dozen different relations for σ . At Mach numbers greater than one, all of these methods give a result larger than the Abramovich model, and thus, poorer agreement with the data. It is somewhat surprising that the Abramovich model predicts the data as well as it does. The results presented in Ref. 13, however, were obtained in studying the base flow behind a projec-

*Since σ appears in the denominator of the expression for G/G_T , a larger σ means that a larger bleed flow is required to produce a given change in plume shape.

IV, B, Modeling Approach and Development (cont.)

tile. Thus, the conditions existing at the dividing streamline stagnation point are different than found in the dual throat nozzle, or in the injector systems studied by Korst and Bauer. A basic difference is that the stagnation point occurs in the axis for the flow behind a projectile, so that axisymmetric effects are critical.

A theoretical determination of conditions that exist in the vicinity of the shear layer attachment stagnation point is one of the remaining unsolved problems in fluid mechanics. Consequently, only semi-empirical methods exist for predicting the pressure of the shear layer. This problem is discussed in the following subsection.

b. Prediction of the Zero Bleed Flow Base Pressure

Three options are available for determining the value to be used for base pressure, P_b/P_0 , for zero bleed flow.

(1) The value can be input directly to the program. If a measured value is available, as for example the values measured in Ref. 10, it should be used.

(2) The value can be calculated automatically by the program. For this option, a pressure is calculated based on the ratio of the area of the secondary nozzle throat to the area of the primary nozzle throat. This value corresponds to the pressure that would be obtained if the nozzle were perfect, i.e., parallel flow downstream of the plume attachment.

(3) A method is also available that calculates pressure ratio by use of a correlation relating the wall angle at plume attachment to a deviation from the area ratio pressure of Method (2), above. The correlation is based on the cold flow results obtained during this program (Ref. 10).

A more general method of determining the base pressure is desirable. This problem is discussed in detail by Roberts, Ref. 12, where the literature on the subject is reviewed and a method is recommended.

c. Calculation of Displacement and Momentum Thickness Across the Shear Layer

The basic definitions for momentum thickness, θ_E , and displacement thickness are

IV, B, Modeling Approach and Development (cont.)

$$\theta_E = \int_{Y_D}^{\infty} \frac{\rho U}{\rho_E U_E} \left(1 - \frac{U}{U_E}\right) dy$$

and

$$\delta^* = \int_{Y_D}^{\infty} \left(1 - \frac{\rho U}{\rho_E U_E}\right) dy$$

where the integrals are taken from the dividing streamline to the interior edge of the shear layer. Since

$$dy = \frac{\chi}{\sigma} d$$

$$U/U_E = \phi$$

and*

$$\rho/\rho_E = \frac{1 - Ca_{\infty}^2}{1 - Ca_{\infty}^2 \phi^2}$$

it follows that

$$\theta_E = \int_{\eta_D}^{\infty} \frac{(1 - Ca_{\infty}^2) (1 - \phi)}{1 - Ca_{\infty}^2 \phi^2} \frac{\chi}{\sigma} d\eta$$

and

$$\delta^* = \int_{\eta_D}^{\infty} \left[1 - \frac{(1 - Ca_{\infty}^2) \phi}{1 - Ca_{\infty}^2 \phi^2}\right] \frac{\chi}{\sigma} d\eta$$

These expressions can be used to obtain the momentum deficit and hence, the thrust loss accumulated in the portion of the shear layer that exits from the nozzle. Note that this calculation also includes the effects of the bleed flow addition since this is accounted for in the shear layer derivation.

*Refer to Appendix A of Ref. 12 for a derivation.

IV, Aerodynamic Mixing Model (cont.)

C. MODEL VERIFICATION

A series of cold flow tests have been conducted by ALRC using dual throat nozzle hardware supplied by the Marshall Space Flight Center. Results of these tests were described earlier in this report.

The aerodynamic model has been used to analyze the nozzle configurations and test conditions included in the cold flow investigation. Hardware configurations for the tests are shown in Figure 36. Test numbers given in Figure 36 are for the test series conducted during Task I of this program. The hardware configuration for the previous IR&D test series was identical with that shown for test 101A-110 of the Task I test series.

Results of each test were presented in Section III. Each set of tests are for a fixed hardware configuration, but with the bleed flow varied. Table VIII gives the separation distance, L/R_{tp} , between the exit plane of the primary nozzle and the throat plane of the secondary nozzle. Also given is the bleed flow rate for each test, and the measured value obtained for the base pressure ratio. This information, together with a definition of the geometry of each nozzle wall, is sufficient to run the aerodynamic model computer program.

The procedure used in carrying out the analysis consisted of the following steps:

(1) The cold flow tests #727-107 through -110 were selected as test cases. The plotted data for these tests are in Figure 37.

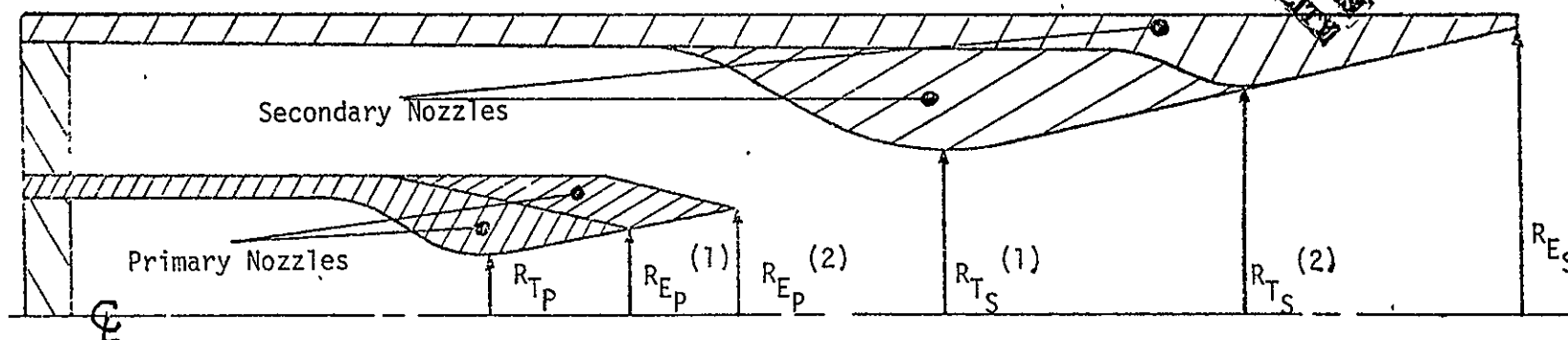
(2) The measured value for base pressure ratio was input to the program. The plume model was then calibrated so that plume attachment was predicted as observed. This required a multiplier of 1.8 for the parameter M_b/γ .

(3) Calculations were made with both the Abramovich shape factor and the Korst shape factor (see Section 3.a. for a discussion of the shape factor, σ). It was found that the Abramovich model was in closest agreement with the cold flow test results, i.e. in predicting the variation of $\dot{W}_S/(\dot{W}_p + \dot{W}_S)$ with P_b/P_0 . Results for two sets of cold flow tests are shown in Figure 38.

(4) Calculations were made for the remaining 34 cold flow tests. The plume model was not recalibrated for these calculations.

Results obtained from the analysis are presented in Table IX, which shows the model predicted percent bleed flow required for plume

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HARDWARE CONFIGURATIONS [cm (in.)]

$$R_{TP} = 1.739 (.6845) \quad R_{ES} = 7.874 (3.100) \quad \epsilon_T = \left(\frac{R_{ES}}{R_{TP}} \right)^2 = 20$$

TEST NUMBER	PRIMARY NOZZLE		SECONDARY NOZZLE	
	R_{EP}	ϵ_P	R_{TS}	ϵ_S
101A - 110	2.244 (.8835)	1.666 ⁽¹⁾	4.562 (1.796)	2.979 ⁽¹⁾
111 - 116	2.244 (.8835)	1.666 ⁽¹⁾	6.175 (2.431)	1.626 ⁽²⁾
117 - 124	3.092 (1.218)	3.164 ⁽²⁾	6.175 (2.431)	1.626 ⁽²⁾
125 - 130	3.092 (1.218)	3.164 ⁽²⁾	4.562 (1.796)	2.979 ⁽¹⁾

Figure 36. MSFC Contractual Test Series, Dual Throat Cold Flow
Mode II Hardware Configurations

TABLE VIII. - COLD FLOW TEST DATA, EACH SET OF TESTS UTILIZE A FIXED
HARDWARE CONFIGURATION

FIGURE NO.	TEST PROGRAM	TEST NUMBER	P_b/P_o^* FOR $W_s = 0$	Le/R_{tp}	$100^* W_s / (W_p + W_s)$, % BLEED FLOW
FIG. 16	IR&D	120,121,123,122	.00576	0	0,3.5,5.7,7.5
FIG. 17	IR&D	101,116,102,103,104	.00883	1.067	0,3.1,4.8,7.9,14.3
FIG. 18	NAS 8-32666	107,108,110,109	.0171	2.883	0,2,3.7,5.5
FIG. 19	NAS 8-32666	102,103,104,105,101	.0229	5.045	0,2,4,5.8,7.7
FIG. 22	NAS 8-32666	112,116,115,114,113,101	.00958	4.013	0,2,3.5,5.7,6.6,8.6
FIG. 23	NAS 8-32666	122,124,123,121	.00415	3.213	0,2.1,4.5,6.2
FIG. 24	NAS 8-32666	118,120,119,117	.00519	3.934	0,2.2,4.5,6.2
FIG. 25	NAS 8-32666	130,126,128,127,125,129	.0162	4.930	0,2.3,4.2,6.1,8,10.2

* MEASURED VALUE

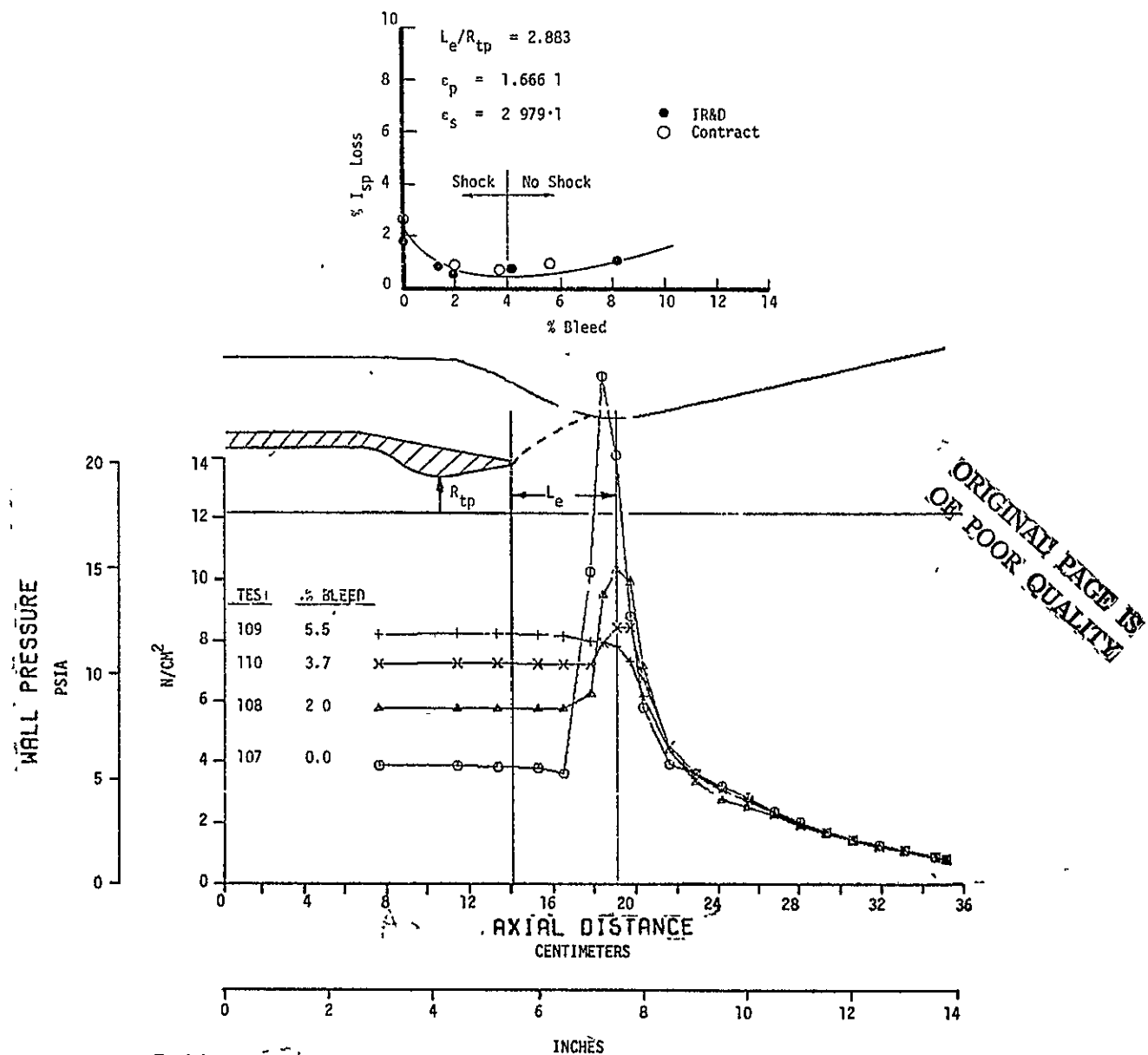


Figure 37. Dual Throat Cold Flow Mode II Plume Attachment Test ($L_e/R_{tp} = 2.883$)

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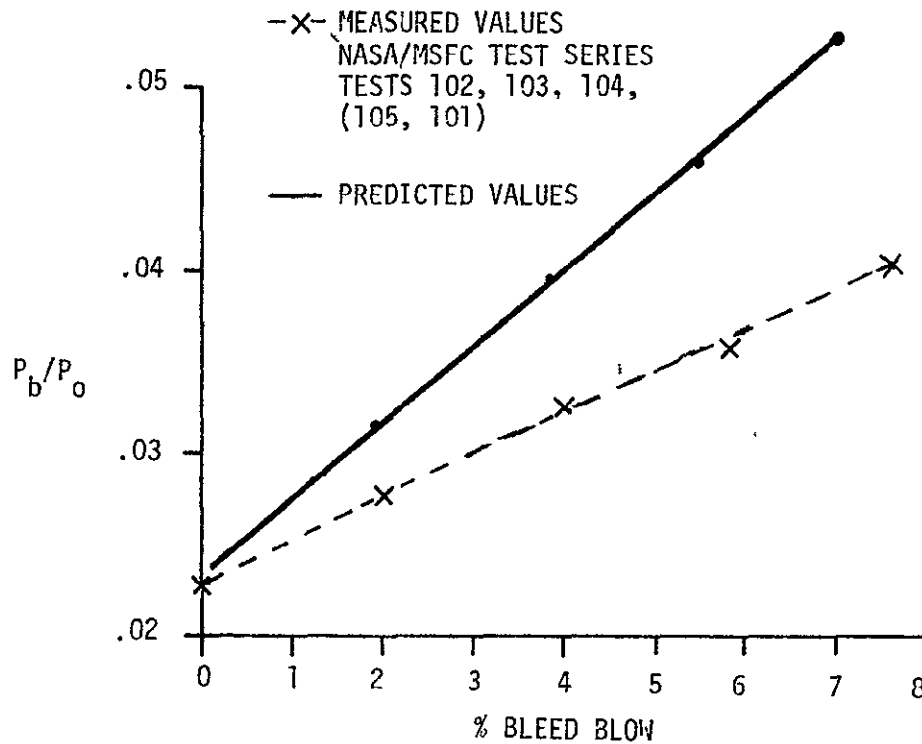
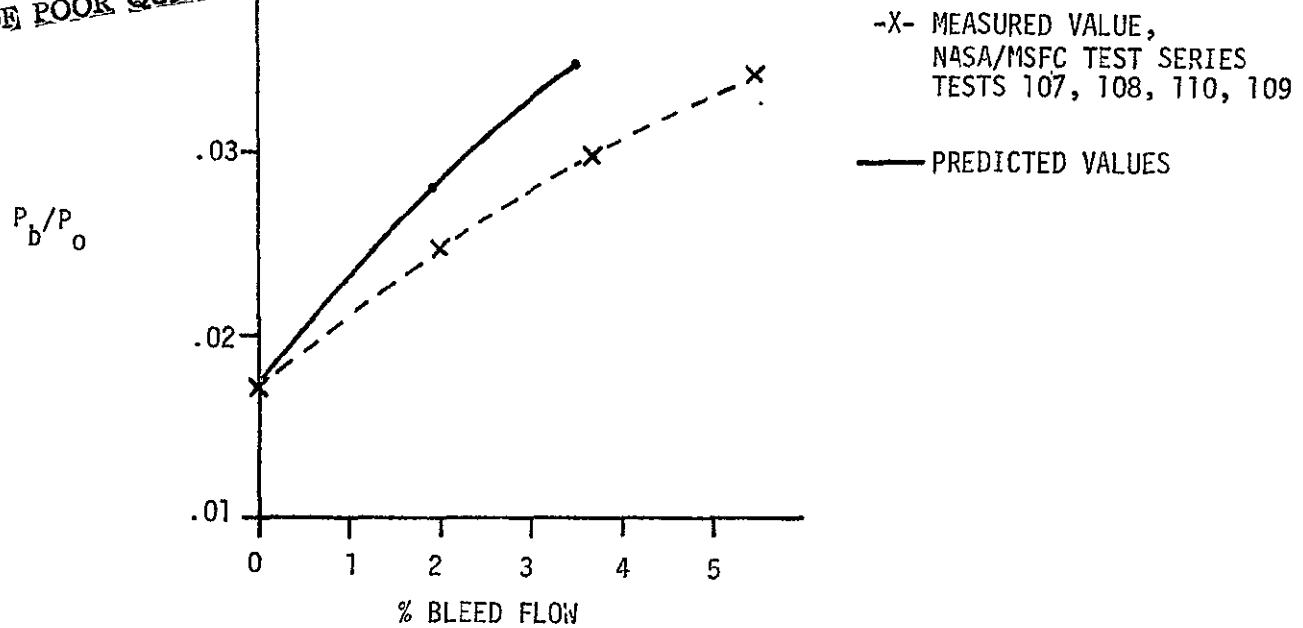


Figure 38. Pressure vs Bleed Flow, Measured and Predicted Values

TABLE IX. - PERCENT BLEED FLOW REQUIRED FOR PLUME ATTACHMENT
AT THE SECONDARY NOZZLE THROAT

TEST CONDITIONS OF FIGURE NO.:	TEST PROGRAM	TEST NUMBER	% BLEED FLOW MEASURED	% BLEED FLOW PREDICTED
FIG. 16	IR&D	N/A	N/A	N/A
FIG. 17	IR&D	101-116	1.5	0.
FIG. 18	NAS 8-32666	110-109	4.6	3.5
FIG. 19	NAS 8-32666	(105,101)	8	7.6
FIG. 22	NAS 8-32666	113-111	7	5.4
FIG. 23	NAS 8-32666	122-124	1	0
FIG. 24	NAS 8-32666	120-119	3	2.8
FIG. 25	NAS 8-32666	128-129	4.5	4.2

IV, C, Model Verification (cont.)

attachment at the nozzle throat. The agreement between this prediction and the cold flow measurements is quite good, which shows that the predicted plume shape varies correctly with the bleed flow rate. Thus a nozzle contour, consisting of the primary nozzle wall, plume shape, and the secondary nozzle wall, is predicted with accuracy.

A MOC calculation was carried out for the design condition predicted for the test case nozzle configuration, i.e. with geometry and test conditions as shown in Figure 37 and with bleed flow sufficient to attach the plume just downstream of the secondary nozzle throat. The effective nozzle geometry for Mode II operation consists of the following:

- (1) the primary nozzle wall
- (2) The plume boundary predicted by the aerodynamic model such that attachment occurs just downstream of the secondary nozzle throat
- (3) the secondary nozzle wall from plume attachment to the exit.

Figure 39 shows a characteristic net calculation for this geometry. The Prandtl-Meyer expansion fan at the primary nozzle exit can be seen in this figure, followed by a region of compression induced by the plume surface. This so called "plume shock" can be seen to form along a right running characteristic surface. At the plume attachment point a shock of greater strength is seen to form due to compression from the secondary nozzle wall. This shock is also formed by the coalescence of right running characteristic surfaces, and then merges with the "plume shock". A region of compression results that is bounded above by the secondary nozzle wall, and below by a right running characteristic. It should be noted that this right running characteristic has a positive slope that is greater than that of the secondary nozzle wall. Thus, most of the flow in the nozzle is unaffected by the shock structure. Shocks are treated by the program as isentropic compression.

Figure 40 shows constant Mach number profiles for the calculation presented as Figure 39. Since the flow calculation is isentropic, these are also surfaces of constant pressure, temperature, and density. It can be seen that the plume surface compresses the flow so that the Mach number is nearly constant along the plume. The divergence efficiency calculated at the exit of the secondary nozzle for this case was found to be 0.981.

Use of these results with the performance methodology developed for Mode II operation is illustrated with a sample calculation presented in Section V of this report.

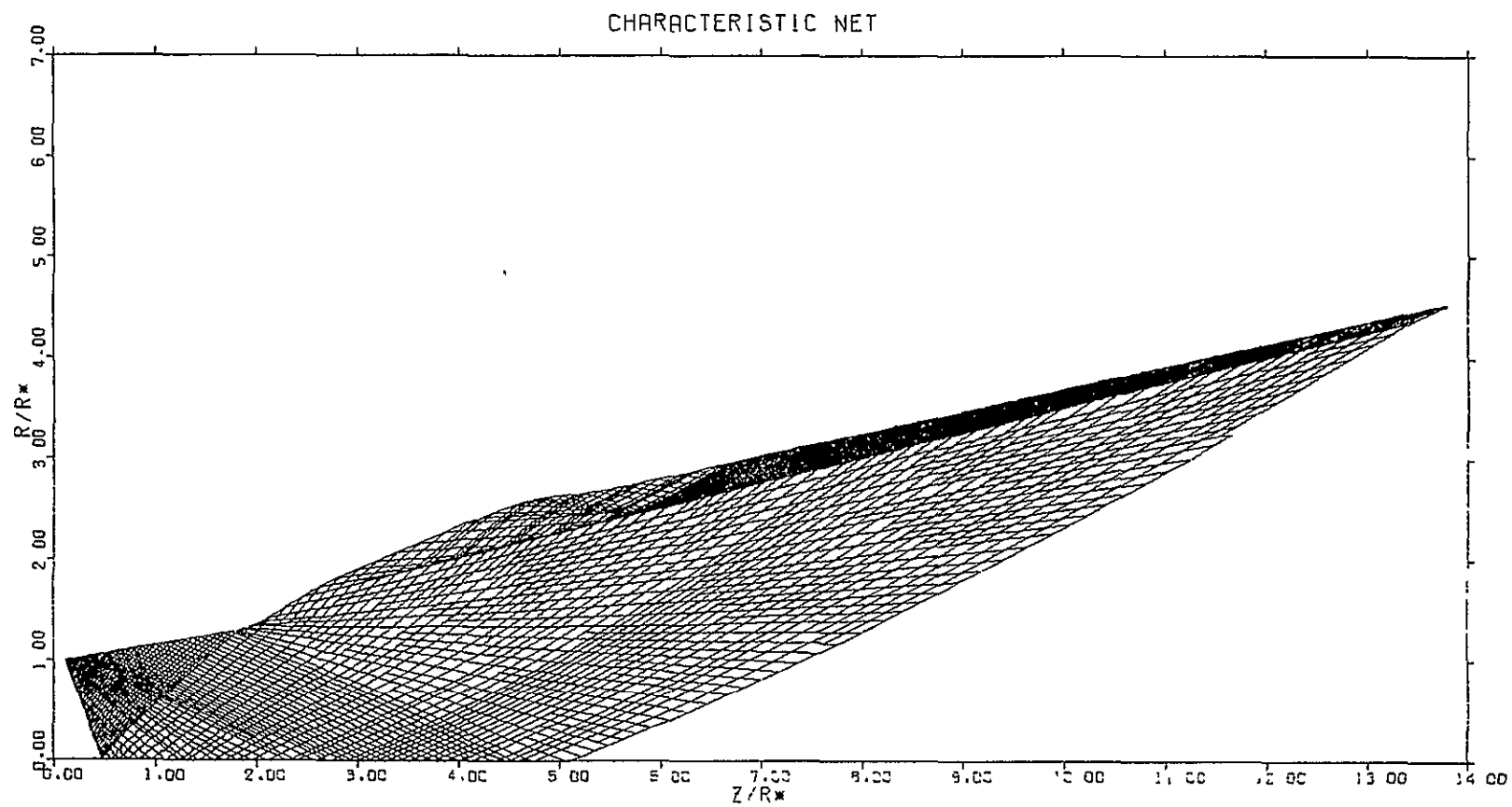


Figure 39. MOC Calculation for the Test Case

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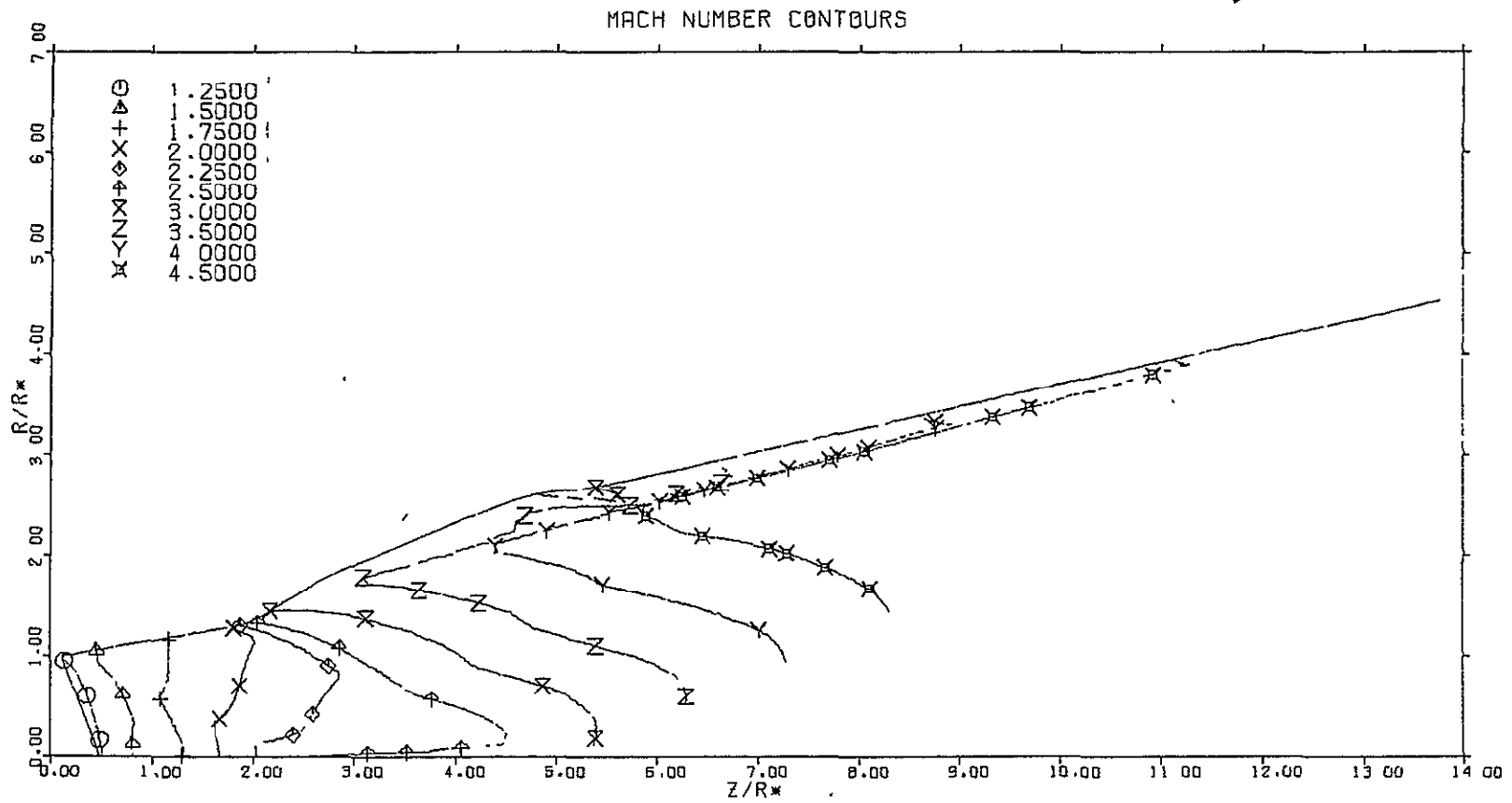


Figure 40. Constant Mach Number Profiles for the Test Case

V. PERFORMANCE METHODOLOGY

A. OBJECTIVES AND GUIDELINES

The objectives of this task were to: (1) formulate a simplified JANNAF performance methodology for the dual throat thruster, and (2) utilize the methodology to predict the hot fire performance of a thruster of similar geometry to the MSFC design.

A simplified performance prediction procedure for liquid propellant thrust chambers has been developed under the sponsorship of the JANNAF Performance Standardization Working Group (Ref. 16). Some of the assumptions in the procedure are specific to expansion nozzles of conventional design. In order to be applicable to the dual throat rocket engine design, the JANNAF procedure requires further development and modification. In the following, a performance methodology is developed for dual throat thrust chambers. This methodology represents an extension of the JANNAF methodology of Reference 16. It is intended to be consistent with the JANNAF methodology so that performance predicted for dual throat engines can be compared with the performance predicted for conventional engines. The computer program analysis tools utilized for conventional engines are also utilized for the dual throat concepts with appropriate input parameters as determined from the aerodynamic model.

B. DUAL THROAT THRUST CHAMBER PERFORMANCE PREDICTION PROCEDURE

1. Method of Approach

The JANNAF performance prediction procedures are concerned with prediction of the specific impulse that will be delivered by a real rocket thrust chamber. The approach used is to first determine an ideal specific impulse, I_{spODE} , for a given rocket thrust chamber. The predicted specific impulse, I_{sppred} , is then calculated by the modeling of known aerodynamic and thermodynamic mechanisms that degrade ideal performance. Ideal performance is dependent upon the physical, chemical, and thermodynamic properties of the propellant combination and its combustion products, and also upon operating conditions such as mixture ratio, chamber pressure, nozzle expansion ratio, and ambient pressure. It is independent of thrust chamber design parameters, such as nozzle geometry, size, material and injector configurations, etc. Thus, one dimensional equilibrium performance, I_{spODE} provides a convenient as well as accurate reference condition for estimating performance efficiency.

Following the procedures described in Reference 16 predicted specific impulse can be related to ideal specific impulse as follows:

$$I_{sp_{pred}} = [\eta_{ER} \cdot \eta_{2D} \cdot \eta_K \cdot I_{spODE}] - \Delta I_{sp_{BL}} \quad (1)$$

V, B, Dual Throat Thrust Chamber Performance Prediction Procedure (cont.)

where the loss models for energy release (ER), two dimensional aerodynamics (2D), and finite rate chemical kinetics (K) are calculated as efficiencies; while boundary layer losses (BL), such as are caused by the nozzle wall boundary layer, are calculated directly.

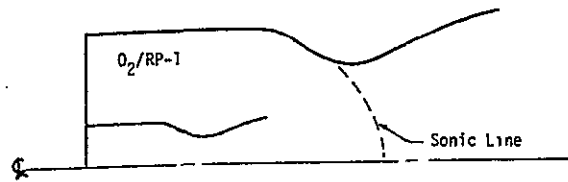
In order to develop the above loss models it is first necessary to establish a control surface bounding the region in which the analysis is to be carried out. This surface is taken as the thrust chamber injector face plus the chamber walls. In References 16, 17, and 18 the assumptions made regarding these loss models are presented and their interrelationships and limitations are fully discussed for rocket thrust chambers of conventional design.

The dual throat chamber concept requires modification and further development of the JANNAF procedure if it is to be applied for performance prediction. The dual throat chamber consists of two conventional thrust chambers, one of which is positioned interior to the other. The interior chamber is referred to as the "primary" chamber, and the exterior is referred to as the "secondary" chamber. The engine is designed to operate in two modes as illustrated in Figure 41. Mode I is a booster mode and itself consists of two modes. In Mode I either the secondary thruster is operated alone (series burn), or with the primary thruster (parallel burn). In the series burn mode (see Figure 41a), the engine is of the conventional type and can be analyzed by the standard JANNAF procedure. The ability to handle this type engine by the procedures developed here allows a direct comparison with the methods given in Reference 16.

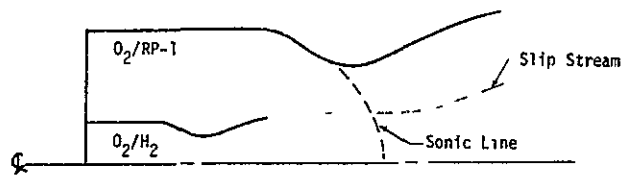
In the parallel burn mode (see Figure 41b), the primary engine is also operating. This mode of operation differs from a conventional engine in that the flow in the primary chamber is expanded supersonically and then shocked subsonically and then expanded through the secondary chamber. A variety of flow situations exist depending on the relative flow rates and stagnation pressures of the two chambers.

The Mode II (sustainer mode) operation is illustrated in Figure 41c. The primary thruster expands the flow to a supersonic condition at the nozzle exit. Further expansion then occurs in the form of a Prandtl-Meyer expansion fan at the nozzle lip producing an exhaust plume with a constant pressure boundary. Flow is introduced into the secondary chamber so as to control the location of the plume boundary in order to minimize reattachment shocks which occur when the flow impinges on the secondary nozzle wall. The plume boundary is an exhaust streamline path and acts very much as a nozzle wall. However, a shear layer develops along this boundary due to viscous interaction of the exhaust jet and the gases recirculating in the primary chamber. Analysis of this shear layer

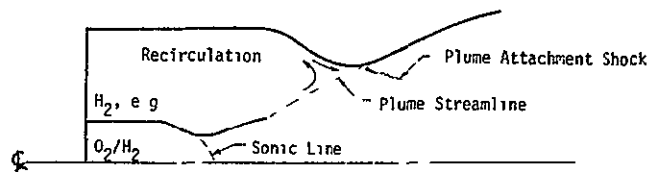
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a. MODE 1 SERIES BURN, BOOSTER MODE



b. MODE 1 PARALLEL BURN, BOOSTER MODE



c. MODE II, SUSTAINER MODE

Figure 41. Dual-Throat Modes of Operation

V, B, Dual Throat Thrust Chamber Performance Prediction Procedure (cont.)

mixing region and the phenomena described above is accomplished with an aerodynamic model developed for the dual throat engine. These results combined with the conventional JANNAF performance procedures are necessary to insure proper performance prediction capability.

2. Performance Prediction Procedure

The dual throat thrust chamber performance procedure is shown in Figure 42. The procedure and each of the loss models are discussed in the subparagraphs below. For the most part this discussion is limited to procedures which differ from those of Reference 16.

The control volume for the thrust chamber has been defined as described previously and it is necessary to determine conditions along this boundary. This is done by standard procedures as is illustrated by the top three rows of Figure 42. Each component of the performance model equation (1), is also illustrated by a row of Figure 42.

a. Procedure for Calculating the Theoretical Specific Impulse, I_{spODE}

The theoretical specific impulse is calculated using the ODE computer program (Ref. 19). The procedure is the same as used in Reference 16. Because the dual throat engine concept implies two flow circuits, the I_{spODE} calculation is also divided into two flow sections or streamtubes. The I_{spODE} for each streamtube is determined independent of the other streamtube. The total engine I_{spODE} is determined by a mass-averaging the two streamtube I_{spODE} values.

b. The Energy Release Loss Model, η_{ER}

The energy release model is based on simplified procedures for estimating losses due to incomplete propellant droplet vaporization and mixture ratio mal-distribution. The generalized length propellant vaporization model (Ref. 20) is suitable for this application. Input to the model includes propellant properties (density, surface tension, heat of vaporization, etc.) and injector jet diameter. The model will determine the amount of propellant vaporization as a function of the chamber shape and length. As such, it is suitable for both a design and analysis tool.

The effect of mixture ratio mal-distribution for both the primary and secondary designs can be approximated by establishing local macroscale stream tubes of different mixture ratios representative of that produced by the injection process. This can be done through the use of cold

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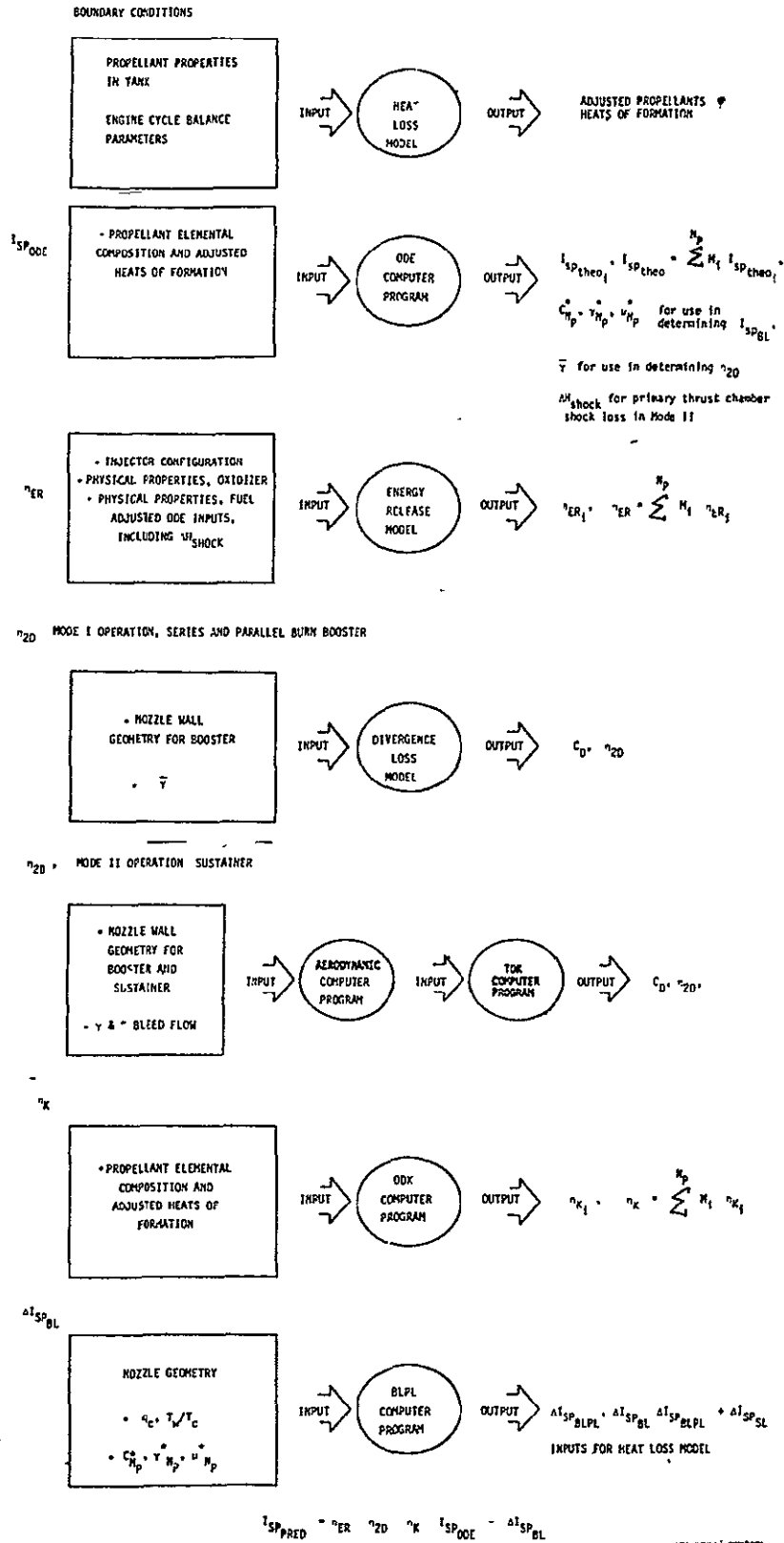


Figure 42. Performance Loss Prediction Procedure

V, B, Dual Throat Thrust Chamber Performance Prediction Procedure (cont.)

flow data from a particular injector concept or analytically from results of the Liquid Injector Spray Pattern (LISP)* (Ref. 21) or previous experience. The stream tubes, once established, are expanded from the chamber inlet to the nozzle exit either one-dimensionally by a simple mass average of the ODE Isp or two-dimensionally using the TDK computer program with multi-stream tubes.

The rate of mixing between the primary and secondary flows achieved during Mode I parallel burn operation can also be approximated using a gas/gas mixing model developed by ALRC during Contract NAS 3-14379 (Ref. 22). The effect of this mixing process can also be approximated by examining the limit cases of no mixing (i.e., two stream tubes) and complete mixing (one stream tube).

c. Film Cooling Losses

If film cooling is required, the associated performance losses are predicted with the energy release model previously described. The flow within the film cooling region is analyzed as a separate flow stream from the main flow stream. These results are then mass averaged to obtain the total engine energy loss for both the main flow and the film cooling flow.

d. The Two Dimensional Loss Model, η_{2D}

The two dimensional loss can be separated into a loss due to upstream nozzle throat curvature and a loss due to nozzle divergence shape.

The loss due to upstream throat curvature affects only nozzle mass flow coefficient, C_D , and, thus, mass flow and C^* . For the flow situations shown in Figure 41, nozzle C_D can be calculated by the following formula

$$C_D = 1 - \left(\frac{\bar{\gamma} + 1}{(1 + R)^2} \right) \left[\frac{1}{96} - \frac{(8\bar{\gamma} - 27)}{2304 (1 + R)} + \frac{754\bar{\gamma}^2 - 757\bar{\gamma} + 3633}{276480 (1 + R)^2} \right] \quad (2)$$

*LISP is a computer program module of the "Distributed Energy Release" (DER) computer program which is used to define liquid spray mass and mixture ratio distribution produced by an array of conventional rocket injector elements.

V, B, Dual Throat Thrust Chamber Performance Prediction Procedure (cont.)

where $\bar{\gamma}$ is averaged to account for flow striations (Equation 2 is relatively insensitive to $\bar{\gamma}$).

The loss due to nozzle divergence shape affects only nozzle thrust. The methods by which this loss may be determined for the flow situations shown in Figure 41 are discussed in the following subsections.

(1) Two Dimensional Loss, Mode I Operation

The simplified procedure of Reference 16 gives charts for determining divergence loss for conical nozzles and optimum nozzles of a Rao type (Ref. 24). These charts are considered sufficiently accurate for evaluating Mode I operation.

(2) Two-Dimensional Loss, Mode II Operation

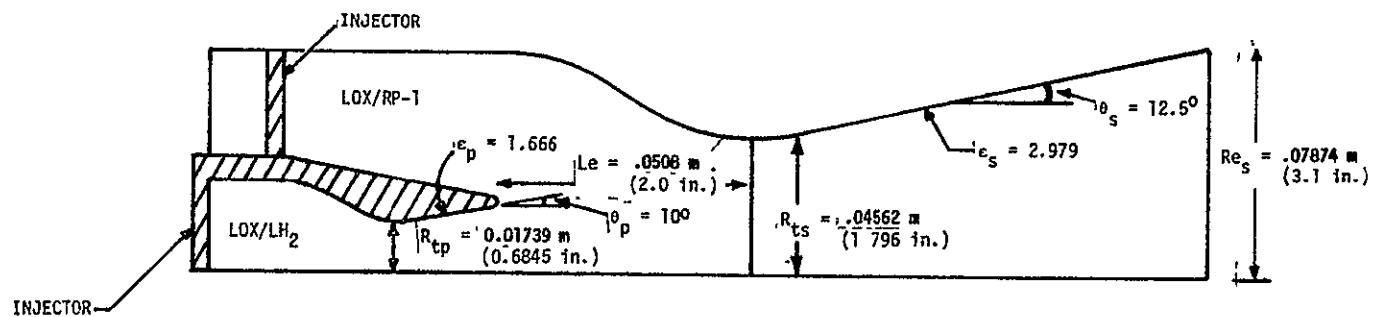
The aerodynamics of the sustainer mode of operation, i.e., Mode II, have been described in detail in Section IV. The output of the aerodynamic computer program is a table of effective nozzle contour points suitable for direct input into the TDK computer program as shown in Figure 42.

The TDK method of characteristics calculation is then performed to determine the effective two-dimensional expansion efficiency for Mode II for expansion through the primary nozzle, along the plume boundary extending from the primary nozzle to the attachment point on the secondary nozzle and finally to the exit of the secondary nozzle. The output from this TDK analysis includes the required η_{2D} for Mode II operation. For this case, shocks are considered as isentropic compressions.

In the event that the above procedure is considered to be of insufficient accuracy, a method of characteristics calculation with an oblique shock capability is required. This more complicated treatment might be required if the nozzle is poorly designed such that plume impingement upstream of the throat gives relatively strong shock. In this case, if the secondary nozzle expansion ratio is large, the simplified TDK calculational path might again be inadequate due to important shock losses.

e. The Chemical Kinetics Loss Model, η_K

The chemical kinetics loss model is identical to that of Reference 16. The high chamber pressures planned for the dual throat engine make this loss negligible except possibly at high area ratio. Several sources of generalized kinetic loss charts and figures exist for LOX/H₂ and LOX/RP-1 propellants which can be used to approximate the kinetic losses.



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Figure 43. Sample Case Geometry

V, B, Dual Throat Thrust Chamber Performance Prediction Procedure (cont.)

A model for each of these systems has been automated in a computer subroutine called CHEM (see Appendix C). An exact solution can also be found for the dual throat configuration through the use of either the ODK or TDK computer models.

f. Boundary Layer Loss Model, η_{BL}

The nozzle wall boundary layer loss is to be calculated using the BLPL computer program developed at the Aerojet Liquid Rocket Company. This computer program is an implementation of the turbulent boundary layer chart procedures given in Reference 17. Input to the program includes operational parameters such as P_c , wall temperature, and γ , design parameters such as R_t , ϵ , and nozzle shape (i.e., conical or bell). The output includes an estimated boundary layer thrust decrement and displacement thickness. Other more rigorous computer solutions also exist including the BLIMP-J and TBL Programs. In addition, the aerodynamic computer model calculates the momentum and displacement thickness growth occurring within the shear layer during Mode II operation. This information can be used with the BLIMP or TBL computer codes to determine the total effective boundary layer loss for Mode II operation. Using this procedure, the effect of the bleed flow on overall performance is accounted for with the shear layer model in terms of an overall boundary layer performance loss.

If required, the transpiration cooling of the nozzle wall produces another loss which must be evaluated. The transpiration cooling mechanism which is modeled as mass injection into the boundary layer is a special case of the boundary layer loss model and the resultant loss caused by mass injection into the boundary can be predicted with the BLIMP-J computer program.

C. DUAL THROAT THRUST CHAMBER PERFORMANCE PREDICTION - SAMPLE CASE

A sample performance prediction case using the procedures described in Section V.B. is presented in this section. The thrust chamber geometry for this sample case was chosen to be identical to the dual throat cold flow test hardware. By choosing the geometry in this manner, both a sample performance calculation and a cold flow to hot fire comparison were made available. The important geometric dimensions are given in Figure 43.

The thrust chamber conditions and the predicted performance values for this sample case are presented in Table X. The one-dimensional equilibrium specific impulse values were obtained using the CHEM computer program described in Appendix C. As indicated in Table X, this performance

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TABLE X. - SAMPLE CASE PERFORMANCE PREDICTION

	MODE I			MODE II	
	PARALLEL		SERIES	PRIMARY	SECONDARY
	PRIMARY	SECONDARY	SECONDARY		
PROPELLANT	LOX/LH ₂	LOX/RP-1	LOX/RP-1	LOX/LH ₂	EXHAUST
Pc, N/cm ² (psia)	1,379 (2,000)	689 (1,000)	689 (1,000)	1,379 (2,000)	80 (116)
O/F	7.0	2.8	2.8	7.	7.
Tc °K (°R)	3,680 (6,624)	3,701 (6,662)	3,701 (6,662)	3,680 (6,624)	3,680 (6,624)
HEAT CAPACITY RATIO, γ	1.18	1.13	1.13	1.18	1.18
C*, m/s (ft/sec)	2,246 (7,370)	1,782 (5,848)	1,782 (5,848)	2,246 (7,370)	2,246 (7,370)
$\dot{W}$ kg/s (LBm/sec)	5.830 12.852	17.957 39.588	28.289 55.752	5,830 12.852	.321 .708
% OF TOTAL	24.5	75.5	100	94.78	5.22
AREA RATIO,	2.979:1	2.979:1	2.979:1	20.5:1	20.5:1
MASS AVERAGED			MASS AVERAGED		
Isp _{ODE} , sec	305.5		287.0	431.0	
η_k	.9931		.9913	.9988	
η_D	.9918		.9918	.9805	.9880*
η_{ER}	.9950		.9950	.9950	
ΔIsp_{BL} , sec	1.8		1.8	7.14	4.62*
Delivered Isp, sec	297.6		279.0	412.8	418.6*
% Isp ODE	97.4		97.2	95.8	97.1*
Thrust, N (Lbf)	69419. (15606.)		69192. (15555.)	24901. (5598.)	25248. (5675.)*

* SIMILAR VALUES FOR CONVENTIONAL, 20:1, 12.5° CONICAL

V, C, Dual Throat Thrust Chamber Performance Prediction - Sample Case (cont.)

from this nonoptimized hardware is about ~97% for Mode I and ~96% for Mode II. The Mode II efficiency is only 1.3% less than that possible from a geometrically similar, conventional engine which compares very well with the approximately 1% loss obtained with the cold flow tests at the same L_e/R_{tp} of 2.883 (Test 727-110).

VI. PRELIMINARY ENGINE SYSTEM MODEL

A. OBJECTIVES AND GUIDELINES

The objective of this task was to define how the analytical results could be incorporated into an engine system model. Since testing was required on the contract to more fully determine the aerodynamic model, the scope of the engine system model task was greatly reduced.

B. ENGINE SYSTEM DEFINITION

A logic chart is given in Figure 44 showing how the results from this contract, combined with related study results, will lead to the definition of a baseline dual throat engine system. The initial step is to establish a preliminary baseline engine (Figure 45 and Table XI) from which parametric data can be generated.

Preliminary designs and parametric data for other engine concepts are available based upon a reference thrust level of 2.70 MN (607,000 lbf) from the Advanced High Pressure Engine Study (NAS 3-19727), and vehicle mission studies were conducted utilizing engines of this thrust level. Therefore, a dual throat engine definition analysis could take advantage of all this prior work if the reference point were taken as 2.70 MN.

The typical preliminary baseline engine given in Table XI has a thrust split $F_1/F_2' = 2.42$ calculated based on the ratio of vacuum thrusts. This split resulted from a selection of a 60% LO₂/RP-1 and a 40% LO₂/LH₂ sea level (Mode I) thrust contribution. This percentage during Mode I operation was found to be optimum in recent studies at NASA/Langley Research Center (Ref. 25) utilizing the dual expander engine. It is expected that the dual throat engine design will optimize at a different point than the dual expander engine. The similarity of the two concepts, however, makes the selected split a good starting point for the conduction of parametric studies.

C. ENGINE SYSTEM MODEL

The major emphasis of this contract has been to generate an aerodynamic model and to incorporate this into a performance model for the dual throat thruster. To obtain an engine system model, the performance model needs to be incorporated as a subroutine with computer models providing cycle power balance, heat transfer/structures/life determination, turbomachinery design, thrust chamber design, nozzle design, component weight scaling, and mission sensitivity. The bulk of this requirement can be achieved in a three task effort as outlined in Figure 44 and given in more detail in Figures 46-48.

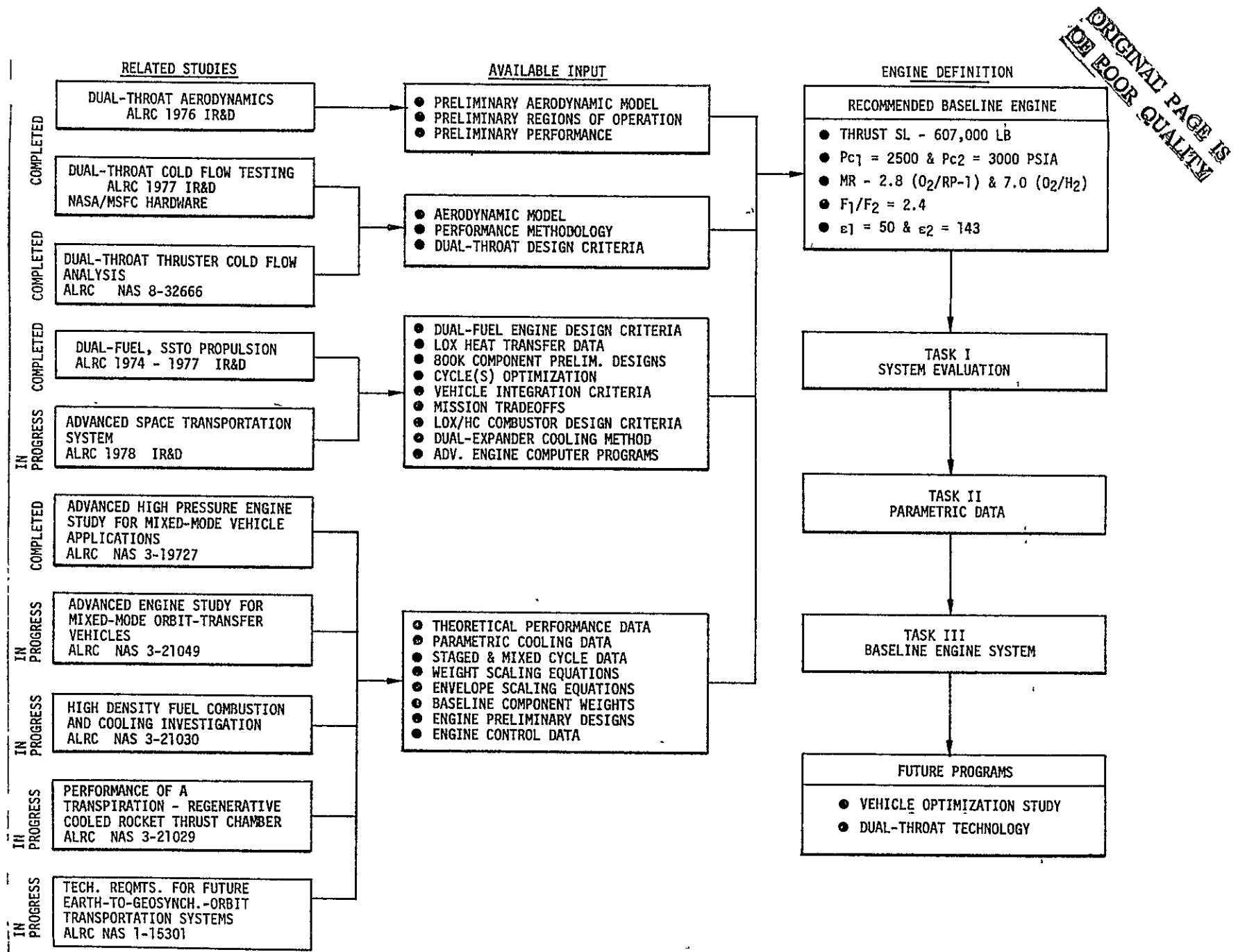
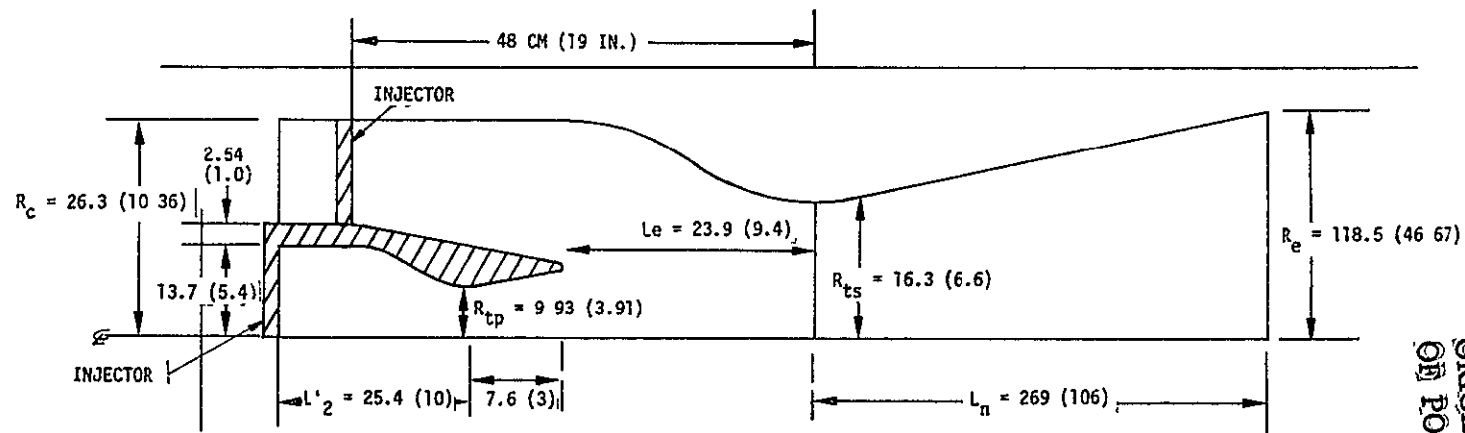


Figure 44. Dual Throat Engine Definition Logic Chart



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Figure 45. Preliminary Baseline Dual Throat Engine

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TABLE XI. - PRELIMINARY BASELINE DUAL-THROAT ENGINE

$$F_1/F_2 = 2.44$$

	60% x 1 LO ₂ /RP-1	40% x 1 LO ₂ /LH ₂	Mode 1 LO ₂ /RP-1 & LH ₂	Mode 2 LO ₂ /LH ₂
Thrust (SL) MN (lb)	1.62 (364,200)	1.08 (242,800)	2.70 (607,000)	-
Thrust (V) MV (lb)	1.90 (427,400)	1.23 (277,300)	3.13 (704,700)	1.28 (288,700)
Mixture Ratio	2.8	7.0	3.63	7.0
Chamber Pressure N/CM ² (psia)	1724 (2500)	2068 (3000)	1724/2068	2068 (3000)
Area Ratio	(50:1)	(50:1)	50:1	142.6:1
ODE I _s (SL) S	310.7	395.9	-	-
ODE I _s (V) S	364.6	452.2	-	470.8
I _s Efficiency %	97	98	97/98	98
I _s (SL) Deliv S	301.4	388.0	330.9	-
I _s (V) Deliv S	353.7	443.2	384.2	461.4
Total Flow Rate Kg/S (lb/S)	548.1 (1208)	283.9 (626)	832.0 (1834)	283.9 (626)
Fuel Flow Rate Kg/S (lb/S)	44.2 (318)	35.5 (78)	179.7 (396)	35.5 (78)
Oxidizer Flow Rate Kg/S lb/S	403.9 (890)	248.4 (548)	652.3 (1438)	248.4 (548)
C* M/S (ft/S)	1803 (5914)	2255 (7399)	-	2255 (7399)
Throat Area CM ² (in. ²)	573 (88.9)	309 (48.0)	883 (137)	309 (48.0)
Exit Diameter CM (in.)	191.0 (75.2)	140.4 (55.3)	237.1 (93.3)	237.1 (93.3)

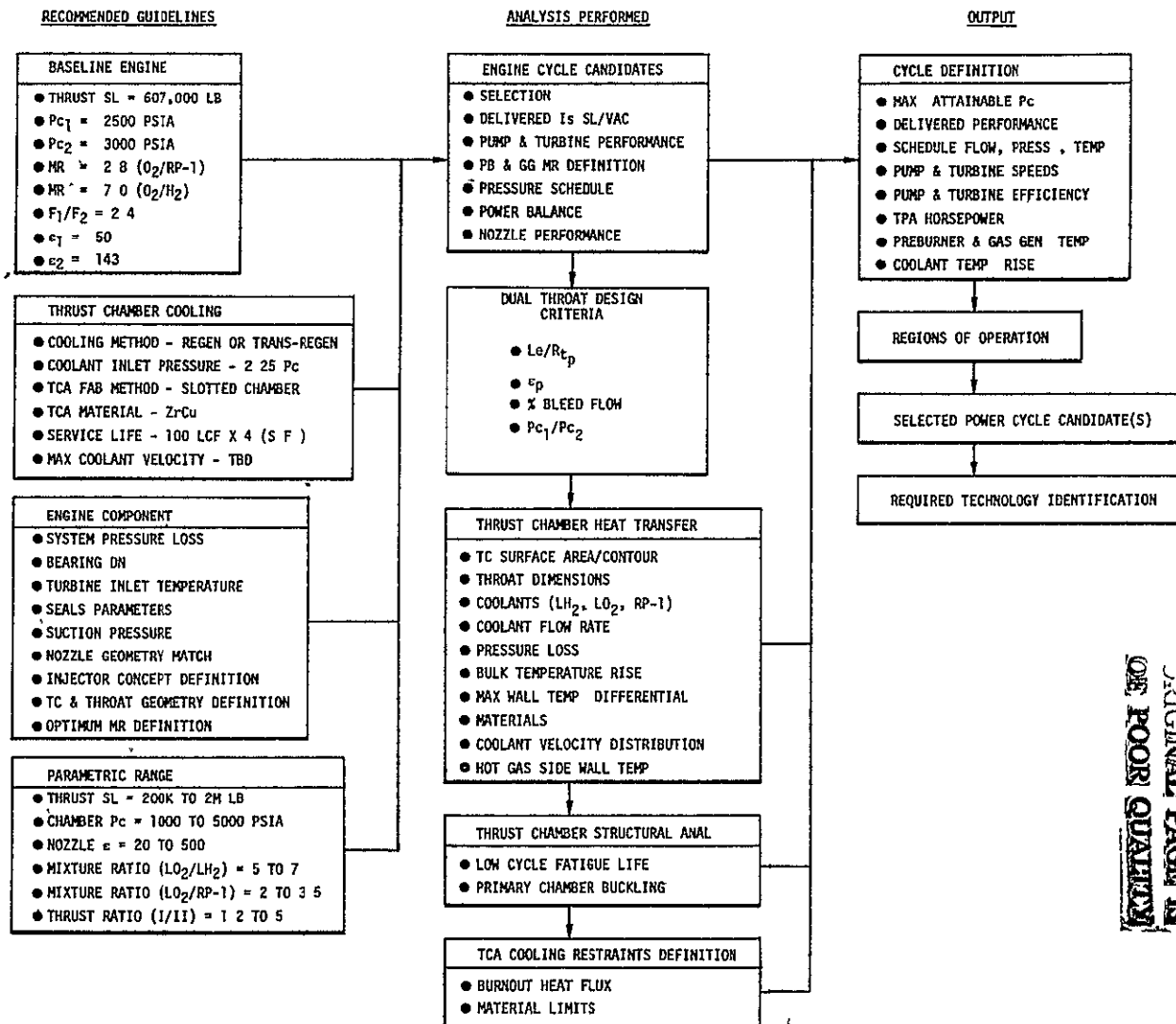
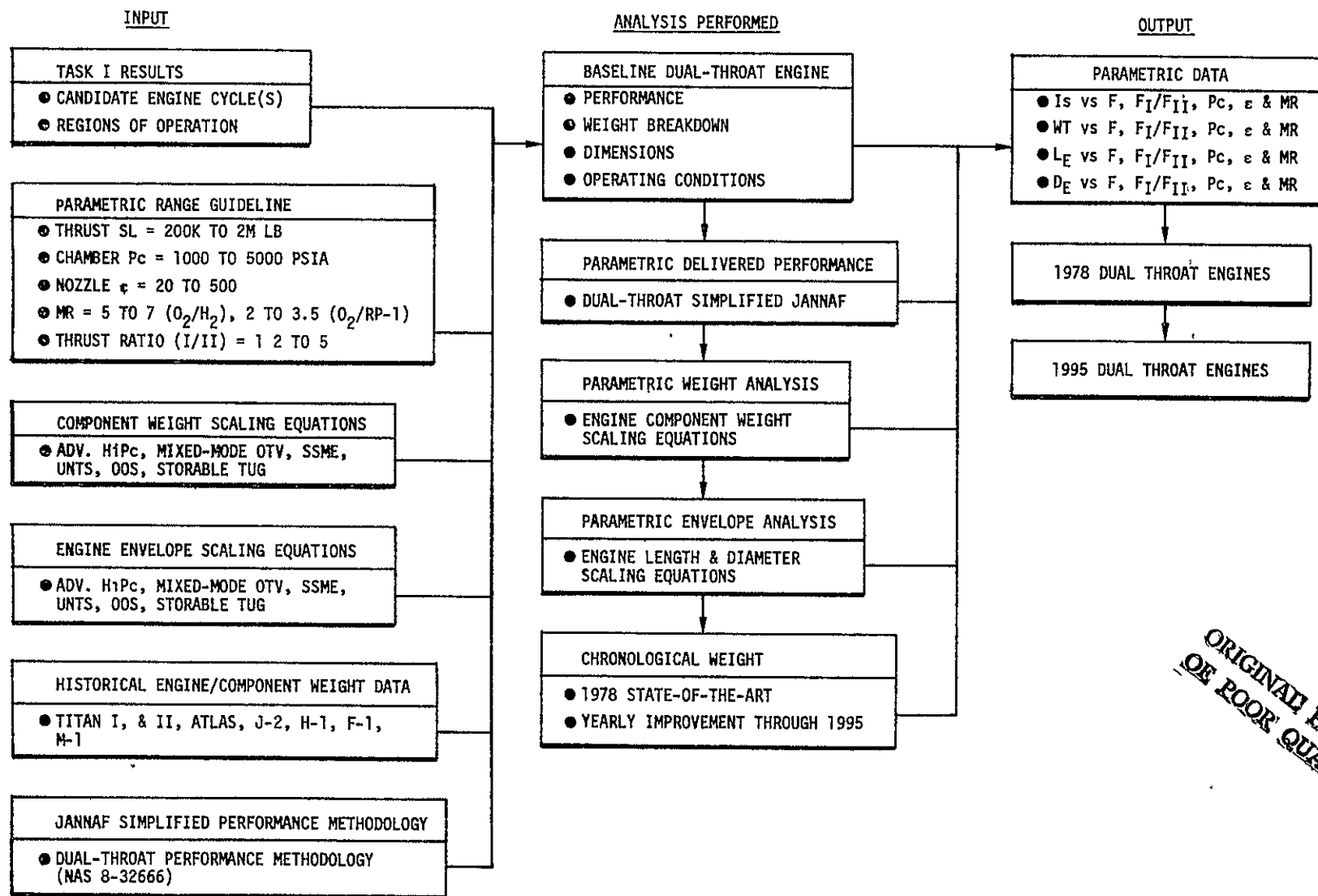


Figure 46. Task I - System Evaluation



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Figure 47. Task II - Parametric Data

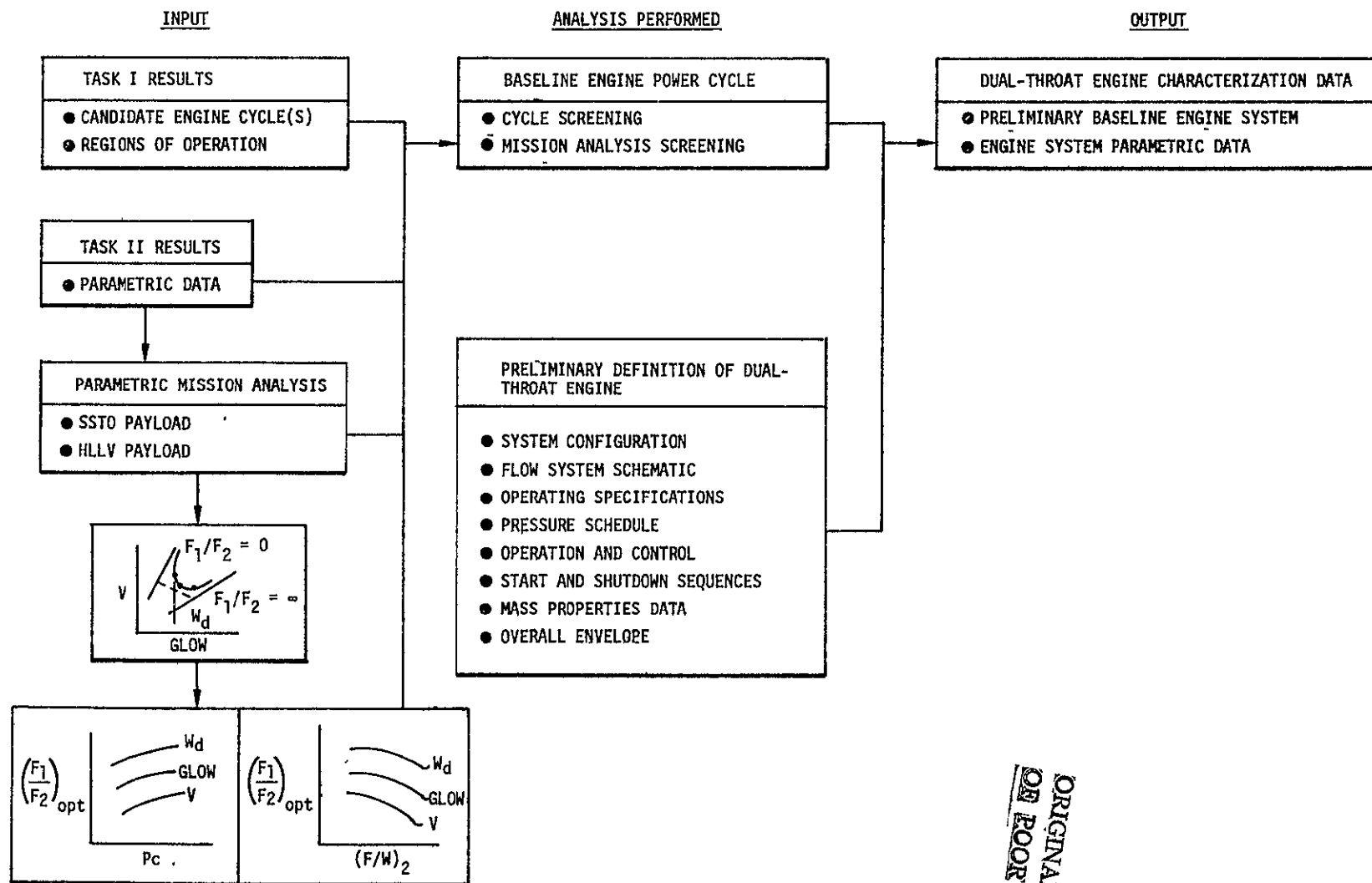


Figure 48. Task III - Baseline Engine System

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VII. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The dual throat cold flow test program has been successfully accomplished with the completion of 69 tests on IR&D and contract funding. The analysis of these data has resulted in a better understanding of the performance trends and characteristics of the dual throat engine.

The cold flow test results coupled with the aerodynamic base flow theories of Korst, Chapman and Bauer form the foundation of the aerodynamic model developed in this program. Verification of the model with the cold flow data gives validity to the analytical techniques used.

The performance prediction methodology developed for the dual throat engine is based on the JANNAF prediction procedures. The aerodynamic model provides the only additional information needed in conjunction with the traditional performance codes and logic given in the JANNAF procedures. Because this methodology is consistent with the JANNAF procedures, a one-to-one comparison between the dual throat and conventional engine performance is possible. Comparisons of this type will prove to be invaluable when determining the advantages and disadvantages of the dual throat design.

B. RECOMMENDATIONS

The aerodynamic model should be modified to predict the base flow pressure using the iterative method given by Roberts, Reference 12. It will probably be necessary to modify the method so that agreement can be obtained with the cold flow data. This method predicts base flow pressure directly as a function of bleed flow, in contrast to the indirect method currently used. Either a direct or indirect method can be used once the zero bleed flow solution has been obtained.

The isoenergetic shear layer model should be modified to treat the mixing of non-isoenergetic streams; i.e., the case where the bleed flow has a different stagnation temperature than the primary flow. The situation where the bleed flow has a different expansion coefficient than the primary flow should also be treated.

Plotting should be added to the program. Plots of interest are:

- (1) wall-plume-wall contours
- (2) base pressure vs bleed flow
- (3) shear layer profiles

VII, B, Recommendations (cont.)

A method should be developed for obtaining two dimensional axially symmetric flow field solutions that include the effect of shock waves. The problem can be important when:

- (1) The operating conditions are off-design so that shock structures develop. It is important to know the effect this can have on performance.
- (2) Mode II operation at design condition will necessarily contain shock waves. Although negligible at low expansion ratio, the shock(s) can become strong downstream. A method should, therefore, be developed for estimating the effect of non-isentropic shock losses on performance. It is also important to be able to predict the nozzle wall pressure so that the possibility of flow separation can be avoided.

Since shock wave development in a nozzle is fundamentally a two-dimensional problem, a two-dimensional computer analysis must be used. It is recommended that the NAP Code (Ref. 14) be applied to one or more nozzle configurations to determine if it is cost effective for this type of problem. A second possibility is to use an existing MOC program that either treats, or can be modified to treat, shock structure. Unlike the NAP analysis, this approach will not work for flows with lambda shock structure.

A series of calculations should be carried out using the aerodynamic model and the performance prediction model that will provide design criteria for the dual throat nozzle optimization. A MOC program would be used to establish optimum nozzle contours for selected designs. Both Mode I and Mode II performance need to be considered. For Mode I, only the secondary nozzle contour is involved and the optimum contour can be determined by the conventional Rao method (Ref. 15). For Mode II, both the primary and secondary nozzle contours are involved. These contours also can be determined using the method of Rao and supersonic start conditions determined from the TDK analysis of the primary nozzle and plume.

For the primary nozzle a suggested approach would be to calculate a family of Rao nozzle contours, each of fixed expansion ratio, but of variable length (i.e., the lip angle constraint would be removed). A contour would be chosen from the results by trading off lip angle vs length.

For the secondary nozzle, a Rao contour would be calculated that is optimum for Mode II. A standard optimum nozzle contour would be generated, i.e., a nozzle that is optimum performing for its length and

VII, B, Recommendations (cont.)

of fixed area ratio. However, the calculation would be started from supersonic conditions calculated by the TDK program using the contour of the primary nozzle, extended by the plume boundary. The result will be the design of two different contours for the secondary nozzle: (1) a nozzle optimized for Mode I operation, and (2) a nozzle optimized for Mode II operation.

Calculating the performance of each nozzle in each mode of operation will allow a trade-off to be made such that an optimum contour can be selected for the mission requirements.

The aerodynamic model should be correlated with hot fire test data. Data at actual engine conditions could be used to verify and fine-tune the aerodynamic model and performance methodology presented herein. Likewise, valuable heat transfer data would be made available during such tests.

The CHEM program described in Appendix C should be upgraded for greater flexibility. The present tables should be extended to include additional propellant combinations (currently LOX/RP-1 and LOX/LH₂ are available). Wider ranges for P_c , area ratio and mixture ratio are also needed to provide adequate coverage of anticipated analyses.

APPENDIX A

AERODYNAMIC MODEL FOR MODE II OPERATION OF THE DUAL THROAT ENGINE

A step-by-step procedure is given below that describes a computer program that models the aerodynamic behavior of the dual throat engine when operating in the sustainer mode, i.e., Mode II. The FORTRAN computer program is called BASE because the aerodynamic model is constructed on a theory for supersonic base flow. The main subprogram for the BASE code is called MAIN and each step discussed below is labeled with comment cards as a separate paragraph in the FORTRAN code for MAIN.

Step 1: Default input values and program constants are set. The program data is read, first a one card table, and then the NAME LIST data, i.e., \$ DATA. The above input is printed. Program constants are calculated from the input and input angles are converted from degrees to radians. The following quantities are then calculated:

$$r_{e_p} = v_p$$

$$r_2^* = r_s / (r_{s_p} r_{e_p})$$

$$x_2^* = (Le/r_p^*)/r_{e_p}$$

Step 2: If IWALL = 2, the primary and secondary nozzle walls are generated. Subroutine WALL is used to calculate vectors:

x , y , and dy/dx

for each nozzle. These vectors are generated in a coordinate centered at the throat of each nozzle and normalized by the throat radius. Next, each vector is transformed into a coordinate system centered at the exit of the primary nozzle and normalized by the primary nozzle exit plane radius. This is called the "plane coordinate system".

The transformations are:

primary nozzle;

$$x_p = (x - x_{e_p})/r_{e_p}, \text{ where } x_{e_p} \text{ is provided by subroutine WALL and corresponds to } y_{e_p}.$$

$$y_p = y \frac{1}{r_{e_p}}$$

$$\left(\frac{dy}{dx}\right)_p = \frac{dy}{dx}$$

secondary nozzle;

$$x_s = x \frac{r^*}{r_p^* r_{ep}} + (L_e/r_p^*) \frac{1}{r_{ep}}$$

$$y_s = y \frac{r^*}{r_p^* r_{ep}}$$

$$\left(\frac{dy}{dx}\right)_s = \frac{dy}{dx}$$

Step 3: Either M_{ep} is input directly, $MEPI \geq 1$; or if $MEPI < 1$, M_{ep} is found as a function of ϵ_p using subroutine MACHS, then:

$$P_{ep}/P_o = \left[1 + \frac{\gamma-1}{2} M_{ep}^2\right]^{-\frac{\gamma}{\gamma-1}}$$

$$T_{ep}/T_o = \left[1 + \frac{\gamma-1}{2} M_{ep}^2\right]^{-1}$$

Step 4: $r_w = r_2^*$

if $(P_b/P_o)_{input} > 0$

then

$$(P_b/P_o) = (P_b/P_o)_{input}$$

$$M_b = \left(\frac{2}{\gamma-1} \left[\left(\frac{P_b}{P_o} \right)^{-\frac{\gamma}{\gamma-1}} - 1 \right] \right)^{1/2}$$

otherwise

$$\epsilon_b = (r_s/r_{sp})^2$$

and

$$M_b = M_b(\epsilon_b) \text{ using subroutine MACHS.}$$

$$(P_b/P_o) = [1 + \frac{\gamma-1}{2} M_b^2]^{-\gamma/(\gamma-1)}$$

$$i = 1$$

Step 5: $(T_b/T_o) = [1 + \frac{\gamma-1}{2} M_b^2]^{-1}$

Values are printed for M_b , P_b/P_i , T_b/T_o

The Crocco Number, Ca_∞^2 , is calculated as

$$Ca_\infty^2 = 1 - T_b/T_o$$

and the following are calculated and printed:

η_D from subroutine ETADS
 $(I_1)_{\eta_D}$ from subroutines IDNES
 $(I_o)_3$ from subroutine IZERDS
 $(I_1)_3$ from subroutine IDNES
 σ from subroutine SIGMAS

Given: γ , M_{e_p} , θ_p

subroutine PLUME is used to find v_b , v_{e_p} , x_j , y_j , y'_j , s_j

subroutine MAX is used to find the index corresponding to the maximum plume radial coordinate. (i.e., The maximum plume length corresponds to this coordinate.)

Step 6: The intersection of the plume x_j , y_j with the secondary nozzle wall is found. This point is called

$$\bar{x}, \bar{y}, \bar{y}', \bar{s}, \text{ and } (\frac{d\bar{y}}{d\bar{x}})_s$$

where $(\bar{x}, \bar{y})$ are the point coordinates, $\bar{y}'$ is the slope of the wall at this point, $\bar{s}$ is the arc length of the plume, and

$(\frac{d\bar{y}}{d\bar{x}})_s$ is the slope of the plume at this point.

Step 7: Mass flow ratio is calculated as

$$(G/G_t)^{(i)} = M_b \left(\frac{2}{\gamma}\right) \left(\frac{1}{r_p^*}\right)^2 \bar{S}r_w \left(\frac{P_b}{P_o}\right)^{\frac{\gamma+1}{2}} 1 - Ca^2 (I_1)_{\eta_D}$$

The percent increase in mass flow over zero bleed flow is calculated as

$$\dot{W}_s^{(i)} / \dot{W}_p = \begin{cases} 0 & \text{for } i = 1 \\ [(G/G_t)^{(i)} - (G/G_t)^{(1)}] * 100 & \text{for } i > 1 \end{cases}$$

Step 8: Values of the intersection point properties calculated in Step 6 and (P_b/P_o) are saved. Next, the pressure is incremented

$$(P_b/P_o) = (P_b/P_o) + \Delta P$$

$$M_b = \left(\frac{2}{\gamma-1}\right) \left[\left(P_b/P_o\right)^{-\frac{\gamma-1}{\gamma}} - 1 \right]^{1/2}$$

$$i = i + 1$$

If any of the following conditions are met, go to Step 9:

$$i > i_{\max}$$

or

$$(P_b/P_o) > (P_{ep}/P_o)$$

or

$$\dot{W}_s^{(i)} / \dot{W}_p > \text{WSWP(NW)}, \text{ which is the largest value requested for output (see \$ DATA).}$$

Next, if IWALL = 1 (i.e., cylindrical wall option) go to Step 5.

If $\bar{x} < x_2^*$ go to Step 5.

Otherwise, proceed to Step 9.

Step 9: Using $\dot{W}_S/\dot{W}_0$ as the independent variable, interpolate in the table saved in Step 8 using as the independent variable $\dot{W}_S/\dot{W}_0 = 0$, WSWP(1), , WSWP(NW) to obtain values for $\bar{x}$, $\bar{y}$, $\bar{y}'$, $(\frac{dy}{dx})_s$, and P_b/P_0 .

Print the following:

$$\dot{W}_S/\dot{W}_p, r_w, z_w, x_w, y_w, \theta_w^\circ, \alpha^\circ, \delta^\circ, P_b/P_0, P_{b_s}, P_{t_2}/P_0$$

where

$\dot{W}_S/\dot{W}_p$ are the input values

$$\left. \begin{aligned} r_w &= \bar{y} \\ z_w &= \bar{x} \end{aligned} \right\} \text{plume/wall intersection in plume coordinates}$$

$$\left. \begin{aligned} y_w &= \bar{y} \frac{r_p^* r_{ep}}{r^*} \\ x_w &= [\bar{x} - (L_e/r_p^*) \frac{1}{\text{rep}}] \frac{r_p^* \text{rep}}{r^*} \end{aligned} \right\} \text{plume/wall intersection in secondary wall coordinates}$$

$$\theta_w^\circ = \frac{180}{\pi} \arctan \left(\frac{dy}{dx} \right)_s, \text{ wall angle at plume/wall intersection}$$

$$\alpha^\circ = \frac{180}{\pi} \arctan \bar{y}', \text{ plume angle at plume/wall intersection}$$

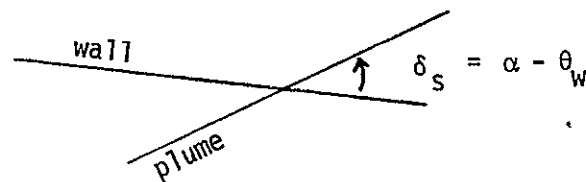
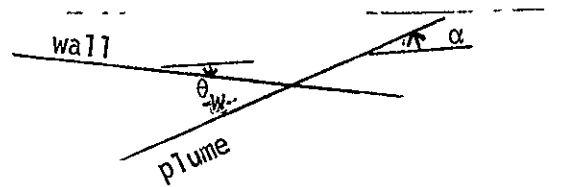
$$\delta^\circ = \alpha^\circ - \theta_w^\circ, \text{ streamline deflection angle corresponding to induced shock}$$

$$P_b/P_0, \text{ pressure ratio along plume boundary}$$

$$P_{b_s} = P_{b_2}/P_0, \text{ pressure ratio behind the shock}$$

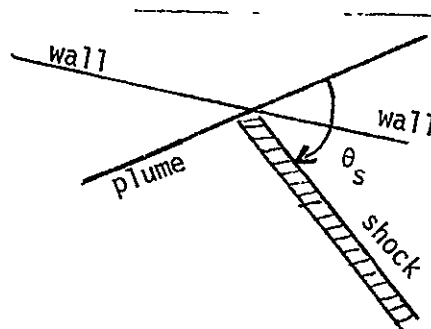
$$(P_{T_2}/P_0), \text{ ratio of total pressure behind the shock to the total pressure before the shock}$$

The angles θ_w° , α° , and δ_s° are shown in the figures below:



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Thus, the plume streamline is deflected through an angle, δ_s . This deflection angle corresponds to a shock angle, θ_s , as shown in the figure below:



Subroutine WAVANG is used to determine the shock angle, θ_s ; the static pressure ratio across the shock, P_2/P_1 ; and the total pressure ratio across the shock, P_{T2}/P_{T1} (same as P_{T2}/P_0 , above).

The above calculations are also carried out and the results printed for the "best solution found". This solution corresponds to the

bleed flow found to be just sufficient to cause the plume to attach just downstream of the secondary nozzle throat; i.e., it is the last solution found as determined by the criteria given in Step 8.

Next, subroutine FIT is used to construct a spline fit of the effective nozzle contour corresponding to the "best solution found". This contour consists of the primary nozzle wall, the plume, and the secondary nozzle wall from the plume attachment to the exit. The coordinate system is that of the primary nozzle. Results are output to be input directly into the TDK computer program.

The momentum thickness, θ_E , and the displacement thickness, δ^* , are calculated at the shear layer cross-section at plume attachment. The integrals are taken from the inside edge of the boundary layer to the dividing streamline; i.e., stagnation streamline, using subroutines IDELTS and ITHETS, respectively. The results are printed.

The program then proceeds to Step 1 to process the next case.

APPENDIX B

PROGRAM INPUT, AERODYNAMIC MODEL

CARD 1:

The 1st card to be input is used to provide a printed heading for the computer output. Columns 2 through 72 are available for text. This card must always precede each case. The remainder of the input data is read using NAMELIST with the name \$DATA, as described below.

\$DATA

Geometry; Option 1, Cylindrical Secondary Wall, see Figure 49.

<u>INPUT ITEM</u>	<u>DEFINITION</u>
IWALL = 1,	The secondary nozzle wall is a cylinder
RS	r^* , radius of the secondary nozzle wall
RSP	r_p^* , radius of the primary nozzle
THETAP	θ_p , exit cone half angle, primary nozzle
EPSP	ϵ_p , expansion ratio of the primary nozzle

The items r^* and r_p^* , above, can be input with units of inches, feet, cm, etc. The input item ϵ_p , above, is used only to determine the Mach number at the exit of the primary nozzle (not required if MEPI is input).

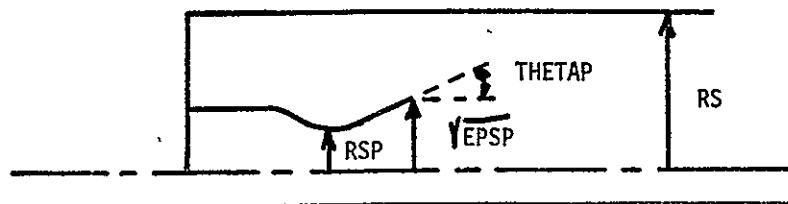


Figure 49. Nozzle Geometry for Option 1

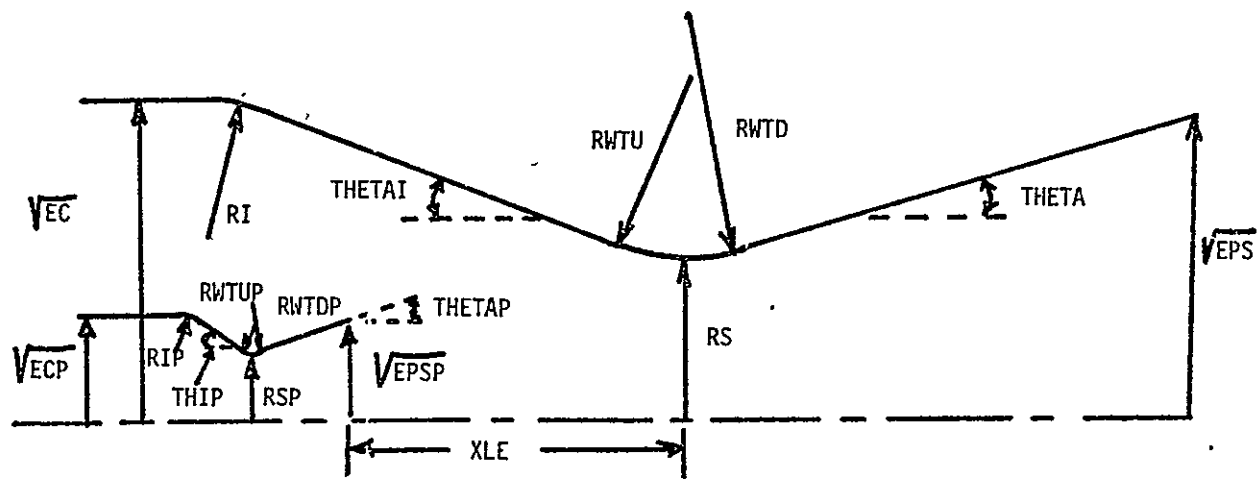


Figure 50. Nozzle Geometry for Option 2

Geometry; Option 2, Full Geometry Option, see Figure 50.

IWALL = 2,	Complete wall geometry specified for both <u>primary and secondary nozzles</u>
<u>Primary Nozzle</u>	
ECP	ϵ_{cp} , primary nozzle chamber contraction ratio
RIP	R_{ip} , wall radius connecting chamber and inlet of primary nozzle
THIP	θ_{ip} , inlet angle for primary nozzle
RWTUP	R_{up} , wall radius on upstream side of the throat, primary nozzle
RSP	r^*_p , throat radius of primary nozzle
RWTDP	R_{dp} , wall radius on downstream side of the throat, primary nozzle
THETAP	θ_p° , exit cone half angle, primary nozzle
EPSP	ϵ_p , expansion ratio of the primary nozzle
<u>Secondary Nozzle</u>	
EC	ϵ_c , secondary nozzle contraction ratio
RI	R_i , wall radius connecting chamber and inlet of primary nozzle
THETAI	θ_i° , inlet angle for secondary nozzle
RWTU	R_u , wall radius on the upstream side of the throat, secondary nozzle
RS	r^* , throat radius of secondary nozzle
RWTD	R_d , wall radius on downstream side of the throat, secondary nozzle
THETA	θ° , exit cone half angle, secondary nozzle
EPS	ϵ , expansion ratio of the secondary nozzle

Geometry; Options 3, 4 or 5. Full Geometry Option with Spline Fit Exhaust Contours -

For the spline fit wall contour options, contour points are input to replace the conical exit cone of either the primary nozzle, secondary nozzle, or both. Other inputs are the same as in geometry Option 2, described above, except that the inputs for ϵ_p , and/or ϵ are not required. The exit cone half angle is interpreted as the spline contour attachment angle. The spline options are as follows:

	<u>Primary Wall</u>	<u>Secondary Wall</u>
IWALL = 3,	Spline	Cone
IWALL = 4,	Cone	Spline
IWALL = 5,	Spline	Spline

The spline coordinates are input as follows:

X4P(2)* =	x_{p_i} , axial coordinates for primary wall spline contour
Y4P(2) =	y_{p_i} , radial coordinates for secondary wall spline contour
N4P =	n_p , $i = 1, 2, \dots n_p$ above. $n_p \leq 20$
TH4P =	θ°_{ep} , spline contour exit angle for primary nozzle, degrees
X4(2) =	x_i , axial coordinates for secondary wall spline contour
Y4(2) =	y_i , radial coordinates for primary wall spline contour
N4 =	n , $i = 1, 2 \dots n$ above. $N \leq 20$.
TH4 =	θ°_e , spline contour exit angle for secondary nozzle, degrees

Relative Positioning of Primary and Secondary
Nozzles (IWALL > 2 Cases Only)

XLE

L_e/r_p^* , axial distance from the exit plane
of the primary nozzle to the throat plane
of the secondary nozzle, normalized by
 r_p^* .

*The first point on a spline contour is automatically calculated by the
program from the exit contour half angle, θ_p° , or θ° , thus each of these
input vectors begins with item 2.

FLOW PROPERTIES AND OPTIONS

<u>INPUT ITEM</u>	<u>DEFINITION</u>	<u>ASSUMED VALUE</u>
DP	Pressure ratio increment for generating bleed flow v.s. pressure table. $P_b^{i+1}/P_o = P_b^i/P_o + DP$	.002
FCTR	Factor for scaling plume shape. Used as a multiplier for the parameter M_b/γ in the Herron method.	1.
G	γ , expansion coefficient	1.4
IPNCH	If IPNCH = 1, geometry inputs for TDK will be punched in the NAMELIST format. If IPNCH = 0, no punched output, only printed output.	0
MEPI	M_{ep} . (REAL type) exit Mach No., primary nozzle. If $M_{Ep} < 1$ then values will be calculated from ϵ_p assuming one-dimensional flow	0
NW	n_w , number of entries in WSWP $n_w \leq 50$	13
OPSIG	Option for Calculating σ If OPSIG = 1, σ from Abramovich If OPSIG = 2, σ from Korst	1
PBPO	Pressure ratio for the plume boundary for zero bleed flow. If not input, the program will calculate a value corresponding to the area of the secondary nozzle throat divided by the area of the primary nozzle throat using the relationship for one-dimensional flow.	0.
WLPRN	If a value other than zero is input, print of the secondary and primary nozzle wall tables will be suppressed (to be used on successive cases when the wall geometry does not change.)	0.
WSWP (1)	$(\dot{W}_s/\dot{W}_p)_i * 100$, for a given value of P_b/P_o percent bleed flow output will be given for these values, $i = 1, \dots, n_w$	0,1,2,3,4,5, 6,7,8,9,10, 12,15

APPENDIX C

THE CHEM COMPUTER PROGRAM

This program calculates equilibrium and kinetic performance values for I_{sp} and C^* . The propellants must be either LOX/RP1 or LOX/H2.

For LOX/RP1 systems the chamber pressure must be between 300 and 1000 psia, the mixture ratio between 2.4 and 3.4, and the nozzle expansion ratio between 1 and 300. Chamber enthalpies for LOX and RP1 are assumed to be -3093 and -6200 calories/mole, respectively.

For LOX/H2 systems the chamber pressure must be between 300 and 2000 psia, the mixture ratio between 5 and 10, and the nozzle expansion ratio between 40 and 3000. Chamber enthalpies for LOX and H2 are assumed to be -3093 and -2154 calories/mole, respectively. Kinetic value for I_{sp} are calculated based on an assumed throat radius of 2.289 inches. These values are corrected as a function of throat radius as follows:

$$I_{sp_{KIN}} = I_{sp_{KIN}} \frac{-\log(2.289/r^*)}{2.289}$$

Program Input

\$DATA

FUEL = The propellant is either LOX/RP1 (=0) or LOX/H2 (=1).

XMAR = Propellant mixture ratio

PC = Chamber pressure, psia

EPS = Nozzle expansion ratio

RT = Nozzle throat radius, inches

\$END

Program Output

The following items are output:

$I_{sp_{ODE}}$, sec

C^*_{KIN} , ft/sec

$I_{sp_{KIN}}$, sec

η_{KIN}

C^*_{ODE} , ft/sec

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